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(NASA-CR-120458) DESIGN, FABRICATION, AND
TEST OF LIGHTWEIGHT SHELL STRUCTURE,
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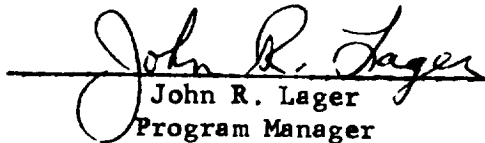
Phase II Quarterly Progress Report No. 2

June 1, 1974 through August 31, 1974

"Design, Fabrication, and Test of
Lightweight Shell Structure"

Contract NAS8-29979

Approved by:


John R. Lager
Program Manager

Prepared for: National Aeronautics and Space Administration
Marshall Space Flight Center
Huntsville, Alabama

Carl Loy
Contracting Officer's Representative

MARTIN MARIETTA CORPORATION
Denver Division
Denver, Colorado 80201

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FOREWORD

This quarterly report was prepared and is submitted by the Denver Division of Martin Marietta Corporation in accordance with the requirements of Exhibit "A", Report Requirements of Contract NAS8-29979. This is an 18 month contract consisting of a 6 month Phase I and a 12 month Phase II. Phase I work was reported in Interim Report No. MCR-74-92, March, 1974. This second quarterly report covers Phase II work performed during the period from June 1, 1974 to August 31, 1974. The program is sponsored by the National Aeronautics and Space Administration, George C. Marshall Space Flight Center, Huntsville, Alabama, with Mr. Carl Loy, the Contracting Officers' Representative (COR). The program is being performed by the Stress, Test, and Advanced Structures Section, Structures and Materials Department, Martin Marietta Corporation--Denver Division, with Mr. John R. Lager serving as Program Manager (PM).

The following Martin Marietta personnel have been principal contributors to the program: Joseph W. Maccalous and Bernard M. Burke, Composite Fabrication; Alan E. Muhl, Metal Fabrication; Arthur Feldman, Materials; Joseph M. Toth, Jr and Alvin Holston, Design and Analysis; and Major L. Sansam and Richard Brown, Structural Test.

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I. INTRODUCTION AND SUMMARY

During Phase I of Contract NAS8-29979, Design, Fabrication, and Test of Lightweight Shell Structure, a cylindrical shell skirt structure 4.57 m (180 in.) in diameter and 3.66 m (144 in.) high was subjected to a design and analysis study using a wide variety of structural materials and concepts. The design loading of 1225.8 N/cm (700 lb/in.) axial compression and 245.2 N/cm (140 lb/in.) torsion is representative of that expected on a typical Space Tug skirt section. Structural concepts evaluated included honeycomb sandwich, truss, isogrid, and skin/stringer/frame. The materials considered included a wide variety of structural metals as well as glass, graphite, and boron-reinforced composites. The most unique characteristic of the candidate designs is that they involve the use of very thin-gage material. Fabrication and structural test of small panels and components representative of many of the candidate designs served to demonstrate proposed fabrication techniques and to verify design and analysis methods. Three of the designs evaluated, honeycomb sandwich with aluminum faceskins, honeycomb sandwich with graphite/epoxy faceskins, and aluminum truss with fiberglass meteoroid protection layers were selected for further evaluation. These concepts result in overall cylinder structural weight in the range 2.59 to 3.08 kg/m² (0.53 to 0.63 lb/ft²). Phase I work was reported in Interim Report No. MCR-74-92, March, 1974.

This second quarterly report covers the second three months effort under Phase II, Fabrication and Test. During this phase, three structural components of each of the three selected structural concepts will be fabricated. A development panel with approximately 1.83 m by 0.915 m (6 ft by 3 ft) overall dimensions will be fabricated for each structural concept. These panels will serve to verify fabrication techniques and will not be subjected to structural test. Successful fabrication of the development panels will be followed by fabrication of 1.83 m by 0.915 m (6 ft by 3 ft) compression panels which will be subjected to axial compression test loading. A 0.915 m by 0.915 m (3 ft by 3 ft) panel of each concept will also be fabricated and subjected to pure shear test loading. In addition, the computer program used to predict the overall buckling of anisotropic cylinders under combined loading is being modified to include cylinders with discrete stringers and frames and theoretical/experimental correlation factors.

Progress during the first quarter of Phase II included, procurement of all materials required for Phase II, structural test plan issued, fabrication drawings completed, fabrication plan completed, fabrication of graphite/epoxy faceskins, chem mill of aluminum faceskins, chem mill of some aluminum truss components and fabrication of graphite/epoxy honeycomb sandwich 1.83 m by 0.915 m (6 ft by 3 ft) development panel.

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Work reported herein for the second quarter of Phase II includes, fabrication of all development and test panels, preliminary panel structural test results, test panel theoretical buckling and strength predictions, preliminary HOLBOAT analysis program modification and fabrication and test of a quality NDE sandwich panel.

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II. PHASE II - FABRICATION AND TEST

Work during Phase II of contract NAS8-29979 involves verification of the predicted potential of three lightweight shell structural concepts designed and selected during Phase I. The aluminum honeycomb sandwich concept utilizes 0.025 cm (0.010 inch) thick 2014-T6 aluminum faceskins bonded to 1.51 cm (0.595 inch) thick 1/8-5052-0.0007-3.1 aluminum hexcel core using 0.0035 inch thick FM-24 film adhesive. The graphite/epoxy honeycomb sandwich concept uses identical core and adhesive but has 0.041 cm (0.016 inch) thick, six layer graphite/epoxy faceskins. The aluminum truss concept uses basic 3.81 cm by 2.86 cm (1 1/2 inch by 1 1/8 inch) 2024-T81 aluminum tubing with 0.125 cm (0.049 inch) wall thickness. These basic tubes are chem milled to different web and flange thicknesses for the individual truss components. The joint attachment is made using doubler plates mechanically fastened with CR-2251 6-2 bulbed cherrylock rivets. A 0.010 cm (0.004 inch) thick fiberglass sheet is bonded to the inner and outer surfaces of the truss to provide meteoroid protection.

Three panels, a 1.83 m by 0.915 m (6 ft by 3 ft) development panel, a 1.83 m by 0.915 m (6 ft by 3 ft) compression test panel, and a 0.915 m by 0.915 m (3 ft by 3 ft) shear test panel will be fabricated for each of the three structural concepts. Successful test of these panels will help to verify the predicted potential of these lightweight shell concepts. Design drawings, fabrication plans and structural test plans for these panels were included in the first quarterly report MCR-74-167, Issue 1.

In addition, aluminum and graphite/epoxy sandwich panels with included defects will be fabricated and subjected to ultrasonic and radiographic NDE to establish defect detection standards.

An analysis effort which was added on to the original contract work during this quarter involves modification of the HOLBOAT cylinder buckling analysis computer program to include discrete stringers and frames and theoretical/experimental correlation factors.

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A. Graphite/Epoxy Honeycomb Sandwich Fabrication

Fabrication of the graphite/epoxy faceskins required for the three sandwich panels was described in the previous quarterly report. Laminate designation, configuration, geometry and weight is summarized in Figure 1.

These faceskin laminates were used in the fabrication of three honeycomb sandwich panels, a development panel (DP-Type I-Gr-16), a compression test panel (CP-Type I-Gr-16) and a shear test panel (SP-Type I-Gr-16). The fabrication drawing for these panels, include fiberglass edge reinforcement for introduction of test loads, is shown in Figure 2. The vacuum bag system used for each sandwich panel is shown schematically in Figure 3. It is important that the aluminum base plate used be very flat since the final cured panel flatness will be highly dependent on tool quality. The layup tool with the top faceskin of a graphite/epoxy panel being put in place is shown in Figure 4. Also shown is the autoclave used to apply pressure and temperature for panel cure. The actual and recommended cure cycle for the panel designated CP-Type I-Gr-16 is shown in Figure 5. The cure panel, without fiberglass end reinforcement, is shown in Figure 6. The fiberglass reinforcement for introduction of test loads was bonded to the cleaned, fully cured panel using room temperature curing epoxy adhesive. The shear test panel, SP-Type I-Gr-16, with fiberglass reinforcement in place is shown in Figure 7. This test panel sustained handling damage as shown in Figure 8 following panel fabrication. The damage consists of a hole in one face-skin, approximately 0.63 cm by 0.63 cm (0.25 in. by 0.25 in.) in size. Local core damage to a depth of approximately 0.25 cm (0.10 in.) was also apparent. This accidental damage provided unscheduled but interesting damage repair information on the lightweight graphite/epoxy panels. The hole was patched with a 2.54 cm (1.0 in) square $\pm 45^\circ$ graphite/epoxy laminate as shown in Figure 8. The area to be patched was cleaned locally and room temperature curing epoxy adhesive applied to the surface and into the fractured area. The patch was then applied and allowed to cure in place under local pressure.

The average weight of the three graphite/epoxy panels fabricated was 2.32 kg/m^2 (0.476 lb/ft^2). This is very close to what was expected based on Phase I small panel development work.

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Faceskin Laminate Designation	Length Inches (Cm.)	Width Inches (Cm.)	Weight, Lb. (Kg.)	Average Thickness Inches (Cm.)
DP-Type I-Gr-16a	73.88 (187.50)	36.37 (92.30)	—	0.018 (6.046)
DP-Type I-Gr-16B	73.88 (187.50)	36.37 (92.30)	—	0.018 (0.046)
CP-Type I-Gr-16a	73.88 (187.50)	36.37 (92.30)	2.70 (1.226)	0.018 (0.046)
CP-Type I-Gr-16b	73.88 (187.50)	36.37 (92.30)	2.78 (1.260)	0.018 (0.046)
SP-Type I-Gr-16	73.77 (187.40)	36.34 (92.40)	2.26 (1.025)	0.015 (0.038)

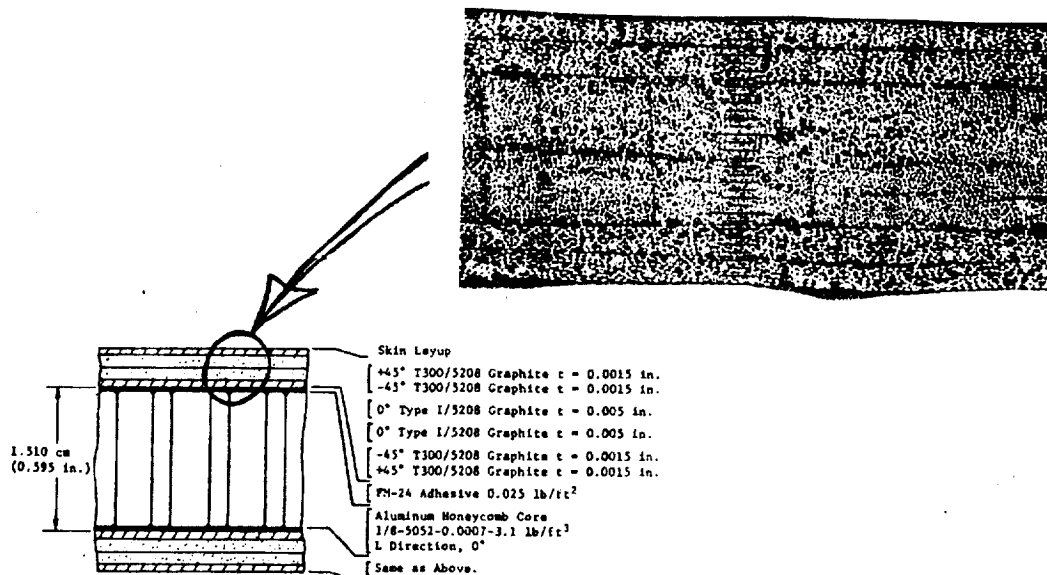
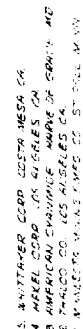
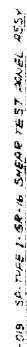


Figure 1

Graphite/Epoxy Faceskin Configuration



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FOLDOUT FRAME

WITHOUT FRAMES

Figure 2
Graphite/Epoxy
Sandwich Panels

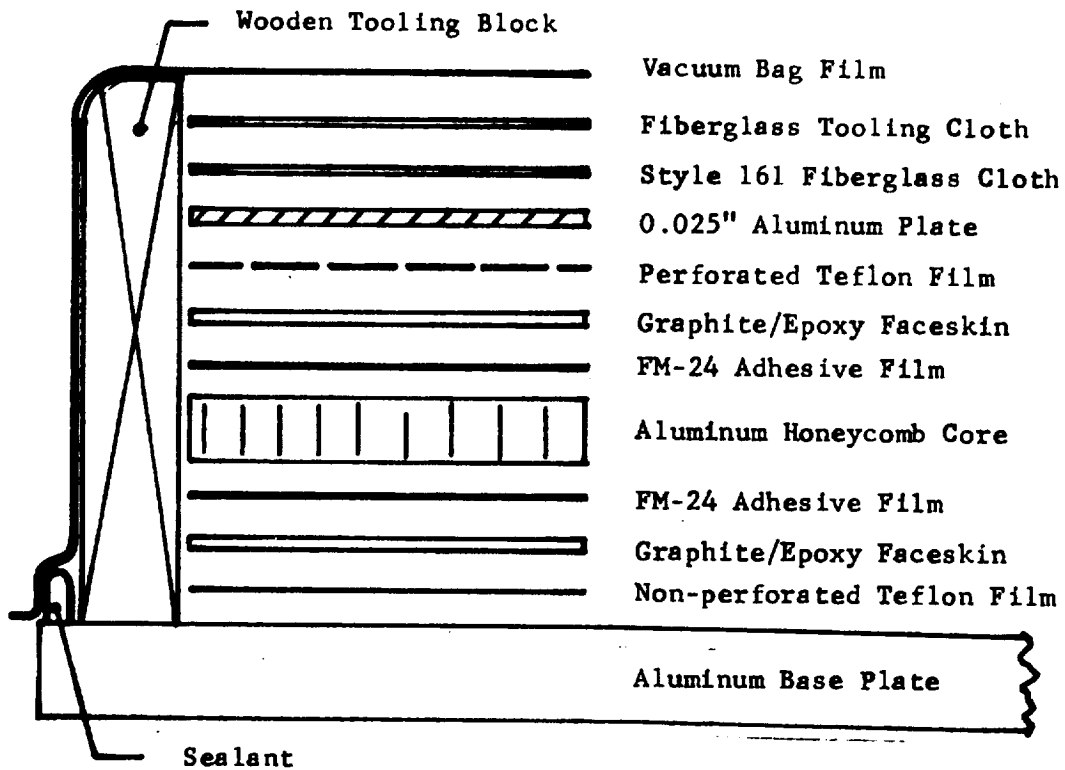


Figure 3
Sandwich Panel
Vacuum Bag System

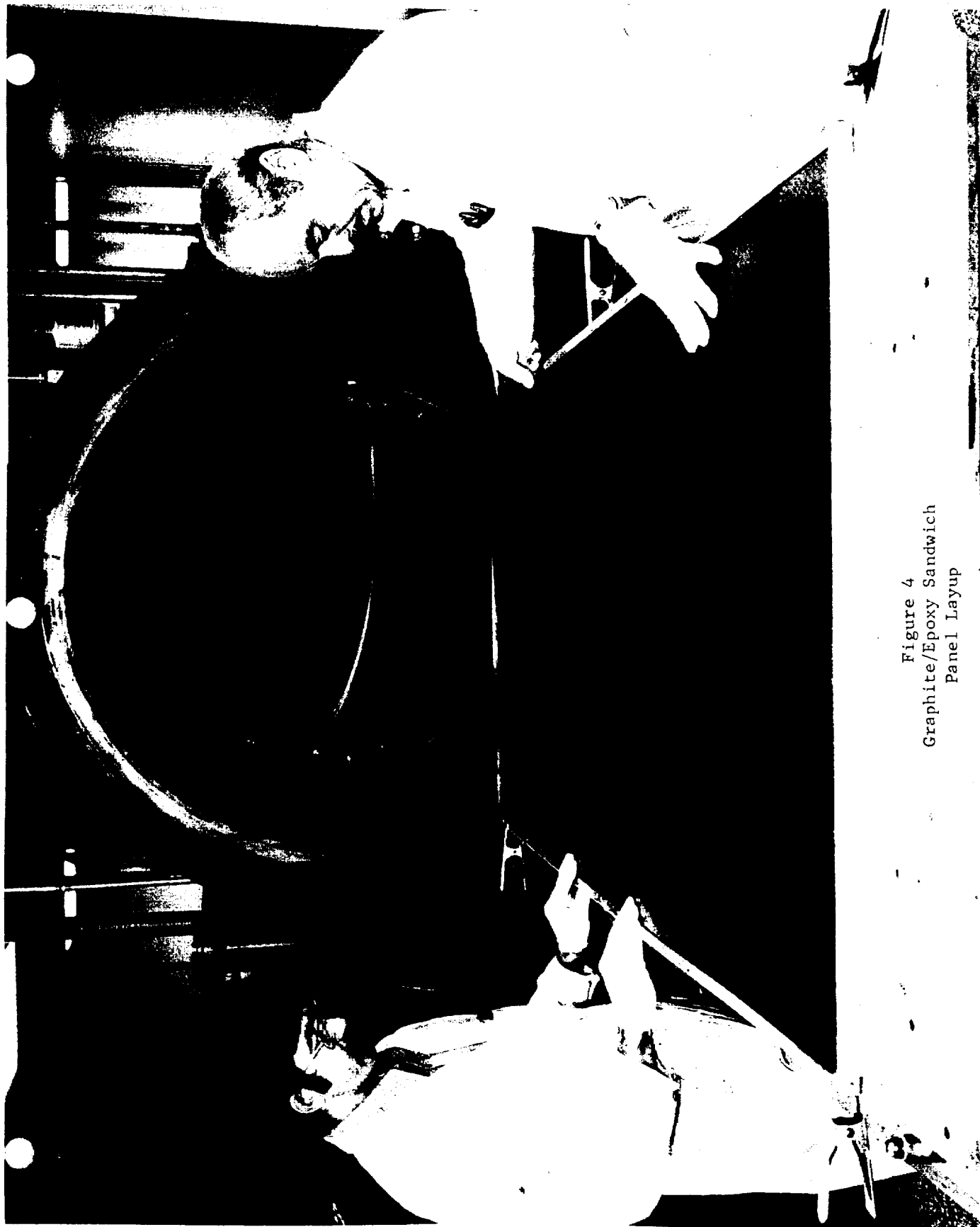


Figure 4
Graphite/Epoxy Sandwich
Panel Layout

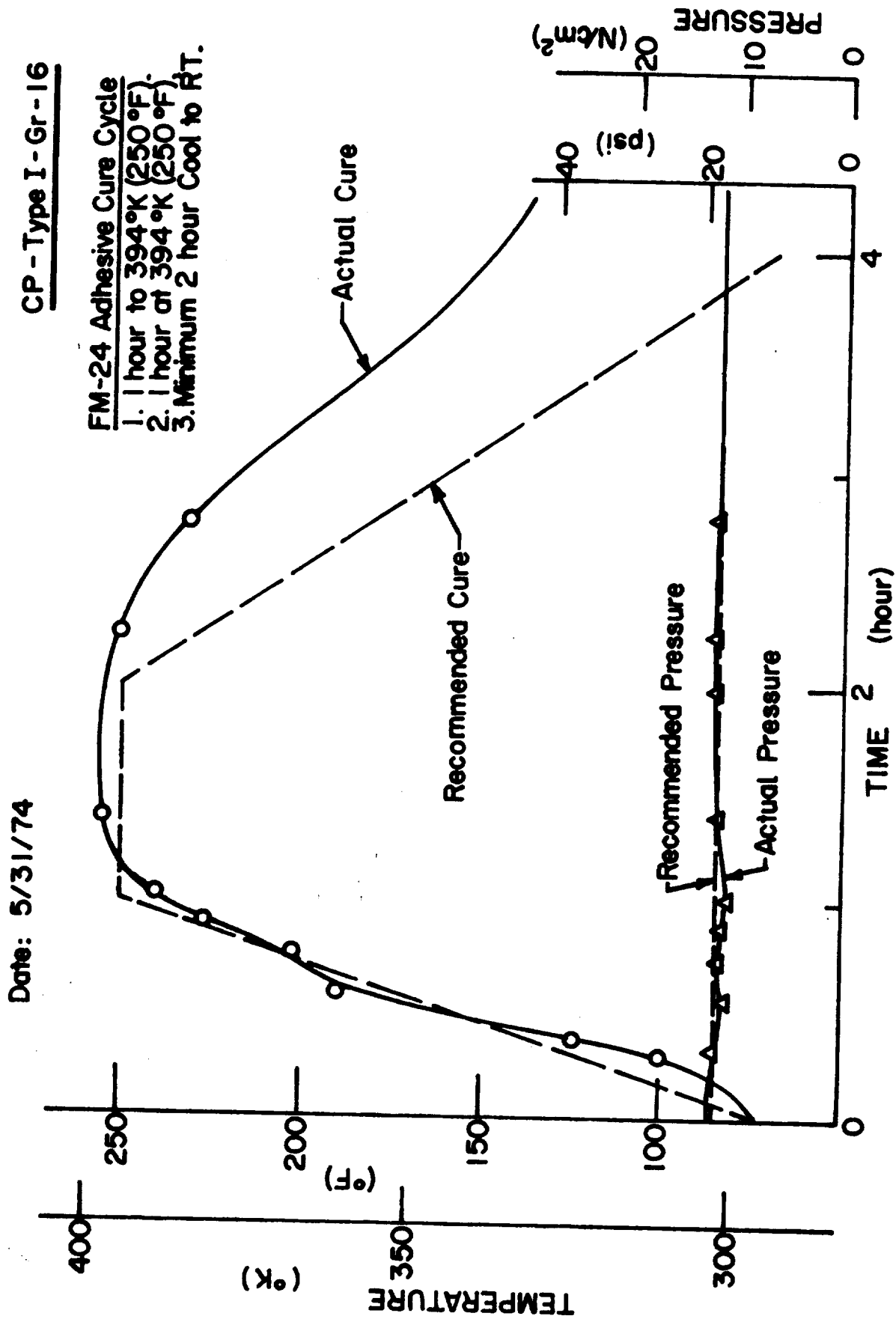


Figure 5
Cure Cycle for
Graphite/Epoxy Sandwich Panel
CP-Type I-Gr-16

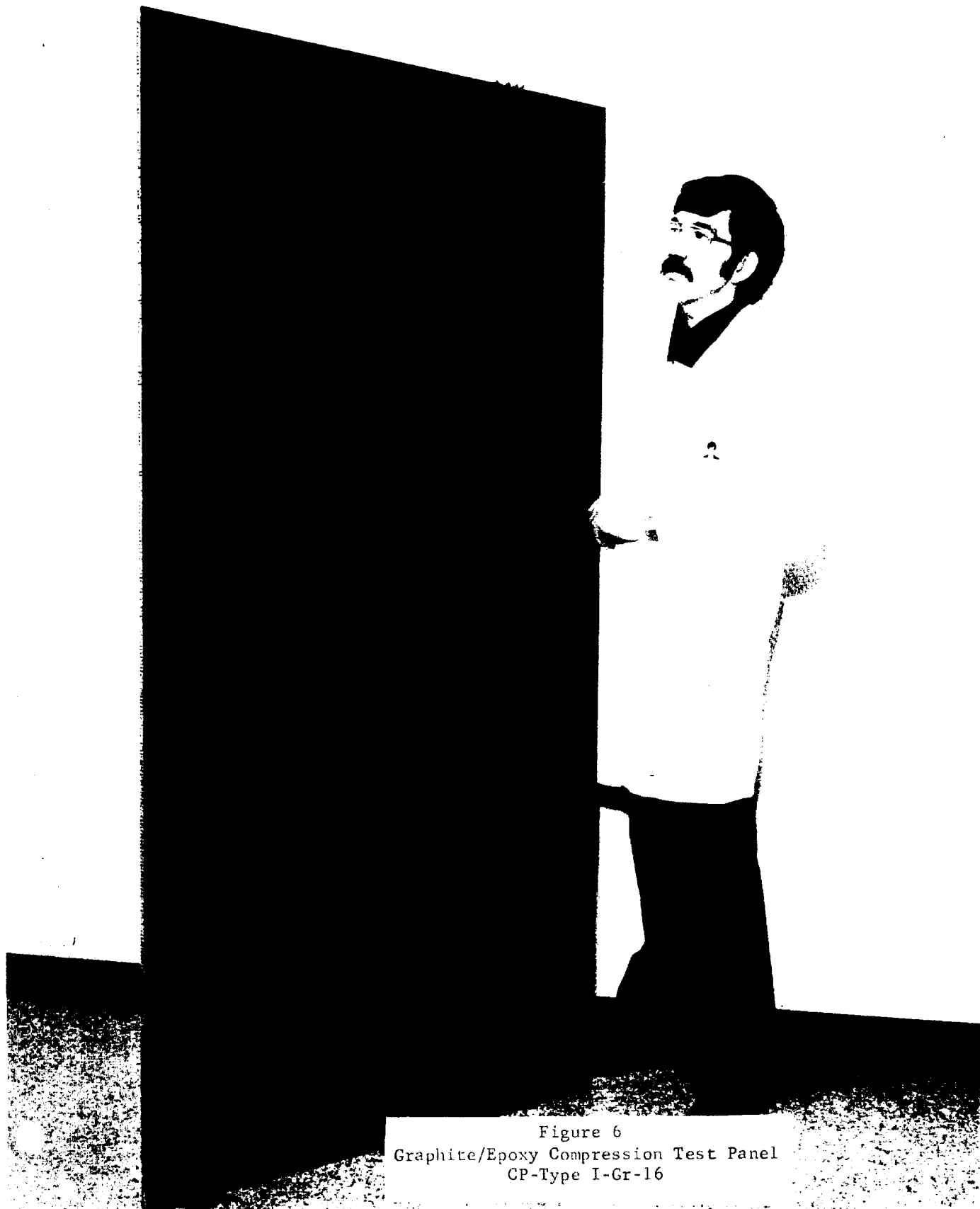




Figure 7
Graphite/Epoxy Shear Test Panel
SP-Type I-Gr-16

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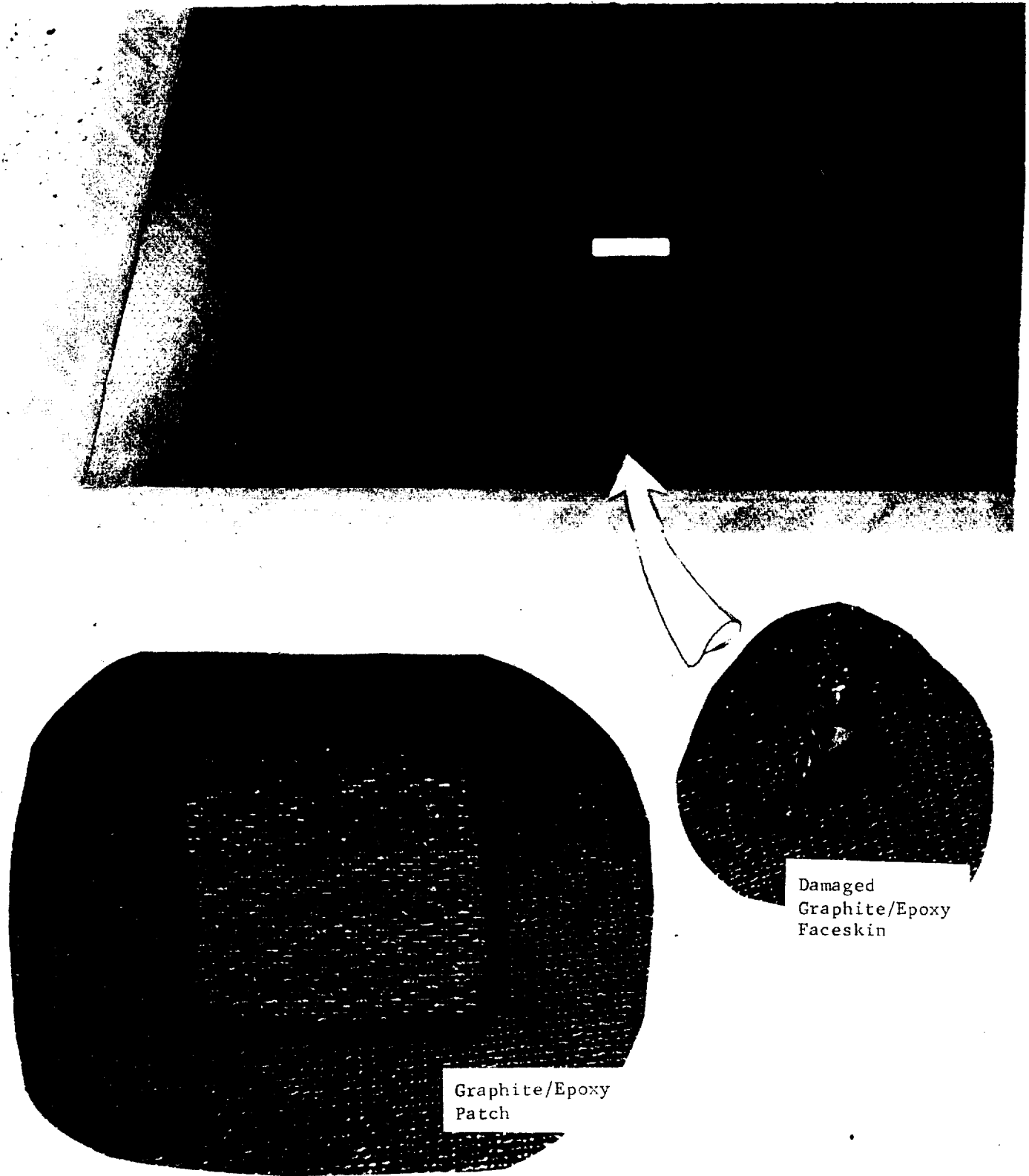


Figure 8
Shear Panel Damage and Repair

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B. Aluminum Honeycomb Sandwich Fabrication

All of the faceskins for the aluminum honeycomb sandwich panels were fabricated during the previous quarter by chemically milling 0.101 cm (0.040 inch) thick 2019-T6 aluminum sheet down to 0.025 cm (0.010 inches) with a tolerance of -0.000 cm. and +0.005 cm. (+0.002 inches) on the finished thickness. Thickness data and comments on the seven 1.83 m by 0.915 m (6 ft. by 3 ft.) aluminum sheets which were chemically milled are listed in Figure 9. Faceskins numbered 2 and 3 were used to fabricate the 1.83 m by 0.915 m (6 ft. by 3 ft.) development panel. These sheets are slightly thicker than originally desired. Experience gained in chemically milling these sheets resulted in the development of techniques required to be able to meet thickness tolerances. Faceskins numbered 4 and 5 with average thickness of 0.0290 cm (0.0114 in) were used to fabricate the (6 ft by 3 ft) panel for compression testing and number 1, with average thickness of 0.0284 cm (0.0112 inches) was used to make the 0.915m by 0.915m (3 ft by 3 ft) shear test panel. The chemically milling techniques developed and the final process were described in the first quarterly report.

Three aluminum sandwich panels, a development panel (DP-ALUM-10), a compression test panel (CP-ALUM-10) and a shear test panel (SP-ALUM-10), shown in Figure 10 were fabricated during this quarter. The development panel was fabricated using the same cure cycle and vacuum bag system as previously described for the graphite/epoxy panels. The cleaning processes used were those found to be satisfactory during Phase I work. The development panel revealed two problems which were subsequently solved. First, the midspan core splice caused a very slight but perceivable local curvature in the upper aluminum skin. This did not happen on previously fabricated graphite/epoxy panels because of their higher local faceskin stiffness. The core splice was eliminated on the compression test panel. A requirement for core splice on large panels would necessitate the use of either thicker faceskins or a local bonded on doubler. Also, the development panel exhibited more overall panel warpage than was considered desirable. This problem was solved by modifying the FM-24 panel cure cycle as shown in Figure 11. The slower heat up to maximum temperature results in reduced thermal gradients and consequently flatter finished panels. The compression test panel and the shear test panel cured using this modified cure cycle. The maximum out of flatness dimension was reduced from 0.152 cm (.060 in) on the development panel to 0.023 cm (0.009 in) on the compression test panel. The average measured weight of the three aluminum sandwich panels, without end attachment capability, was 2.52 Kg/in² (0.516 lb/ft²). This is a typical panel weight for 0.025 cm (0.010") minimum gage aluminum and ± 0.0025 cm (± 0.001 in.) chem mill tolerance. The finished compression and shear test panels are shown in Figures 12 and 13 respectively.

Sheet Number	Minimum Thickness, Inches (cm.)	Maximum Thickness, Inches (cm.)	Average Thickness, Inches (cm.)	Comments
1	0.0105 (0.0267)	0.0119 (0.0302)	0.0112 (0.0284)	Good
2	0.0115 (0.0292)	0.0134 (0.0340)	0.0121 (0.0307)	Slightly Thick
3	0.0112 (0.0284)	0.0132 (0.0335)	0.0121 (0.0307)	Slightly Thick
4	0.0106 (0.0269)	0.0118 (0.0300)	0.0114 (0.0290)	Good
5	0.0109 (0.0277)	0.0121 (0.0307)	0.0114 (0.0290)	Good
6	0.0098 (0.0249)	0.0115 (0.0302)	0.0110 (0.0279)	One small Wrinkle
7	0.0101 (0.0256)	0.0120 (0.0305)	0.0111 (0.0282)	Two small Wrinkles

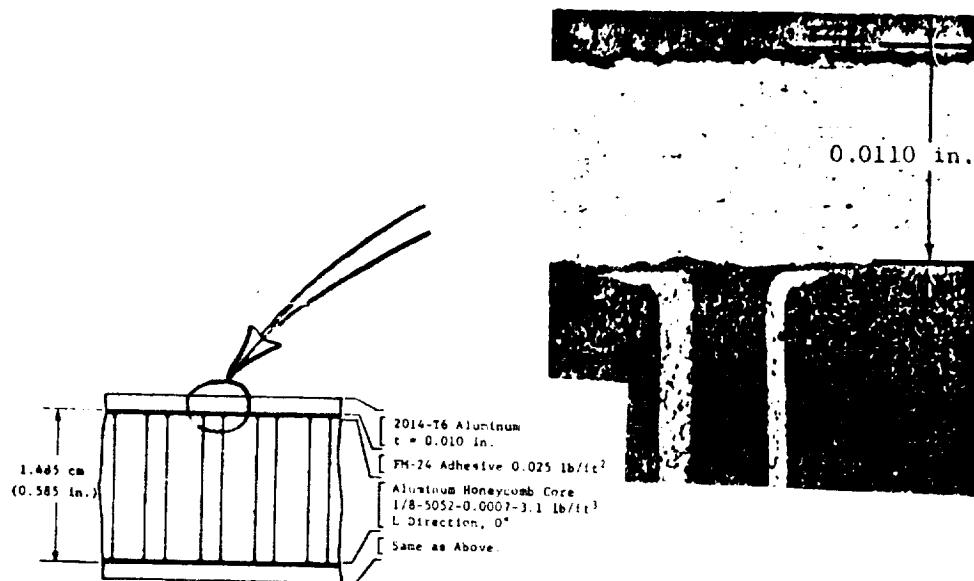


Figure 9
Aluminum Faceskin Thicknesses

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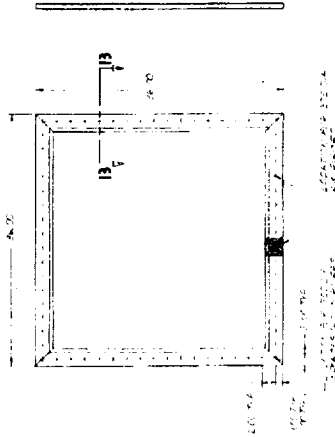
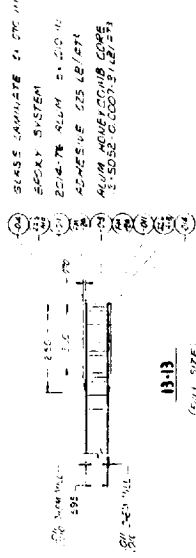
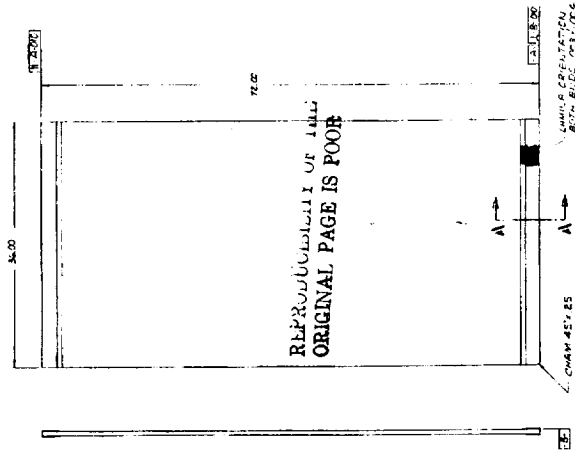
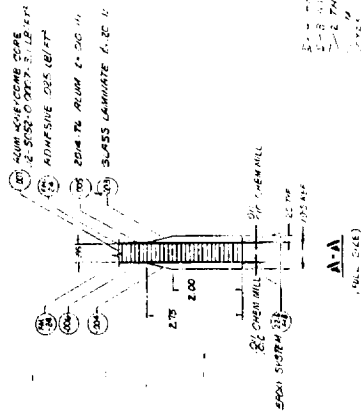


Figure 10
Aluminum Sandwich Panels



008 CP ALUM-10 COMPRESSION TEST PANEL ASSY
010 DP ALUM-10 DEFLACEMENT PANEL ASSY



FOLDOUT FRAME

FOLDOUT FRAME

ITEM	DESCRIPTION	QTY	UNIT	PRICE	TOTAL
1	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
2	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
3	ALUM. HOUSING CORE	1	PC	1.00	1.00
4	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
5	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
6	ALUM. HOUSING CORE	1	PC	1.00	1.00
7	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
8	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
9	ALUM. HOUSING CORE	1	PC	1.00	1.00
10	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
11	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
12	ALUM. HOUSING CORE	1	PC	1.00	1.00
13	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
14	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
15	ALUM. HOUSING CORE	1	PC	1.00	1.00
16	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
17	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
18	ALUM. HOUSING CORE	1	PC	1.00	1.00
19	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
20	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
21	ALUM. HOUSING CORE	1	PC	1.00	1.00
22	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
23	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
24	ALUM. HOUSING CORE	1	PC	1.00	1.00
25	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
26	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
27	ALUM. HOUSING CORE	1	PC	1.00	1.00
28	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
29	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
30	ALUM. HOUSING CORE	1	PC	1.00	1.00
31	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
32	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
33	ALUM. HOUSING CORE	1	PC	1.00	1.00
34	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
35	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
36	ALUM. HOUSING CORE	1	PC	1.00	1.00
37	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
38	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
39	ALUM. HOUSING CORE	1	PC	1.00	1.00
40	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
41	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
42	ALUM. HOUSING CORE	1	PC	1.00	1.00
43	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
44	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
45	ALUM. HOUSING CORE	1	PC	1.00	1.00
46	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
47	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
48	ALUM. HOUSING CORE	1	PC	1.00	1.00
49	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
50	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
51	ALUM. HOUSING CORE	1	PC	1.00	1.00
52	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
53	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
54	ALUM. HOUSING CORE	1	PC	1.00	1.00
55	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
56	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
57	ALUM. HOUSING CORE	1	PC	1.00	1.00
58	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
59	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
60	ALUM. HOUSING CORE	1	PC	1.00	1.00
61	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
62	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
63	ALUM. HOUSING CORE	1	PC	1.00	1.00
64	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
65	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
66	ALUM. HOUSING CORE	1	PC	1.00	1.00
67	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
68	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
69	ALUM. HOUSING CORE	1	PC	1.00	1.00
70	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
71	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
72	ALUM. HOUSING CORE	1	PC	1.00	1.00
73	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
74	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
75	ALUM. HOUSING CORE	1	PC	1.00	1.00
76	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
77	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
78	ALUM. HOUSING CORE	1	PC	1.00	1.00
79	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
80	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
81	ALUM. HOUSING CORE	1	PC	1.00	1.00
82	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
83	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
84	ALUM. HOUSING CORE	1	PC	1.00	1.00
85	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
86	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
87	ALUM. HOUSING CORE	1	PC	1.00	1.00
88	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
89	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
90	ALUM. HOUSING CORE	1	PC	1.00	1.00
91	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
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93	ALUM. HOUSING CORE	1	PC	1.00	1.00
94	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
95	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
96	ALUM. HOUSING CORE	1	PC	1.00	1.00
97	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50
98	ADHESIVE 125 LB/ET	1	ET	1.00	1.00
99	ALUM. HOUSING CORE	1	PC	1.00	1.00
100	GLASS LAMINATE 1/8" THICK	1	SQ YD	1.50	1.50

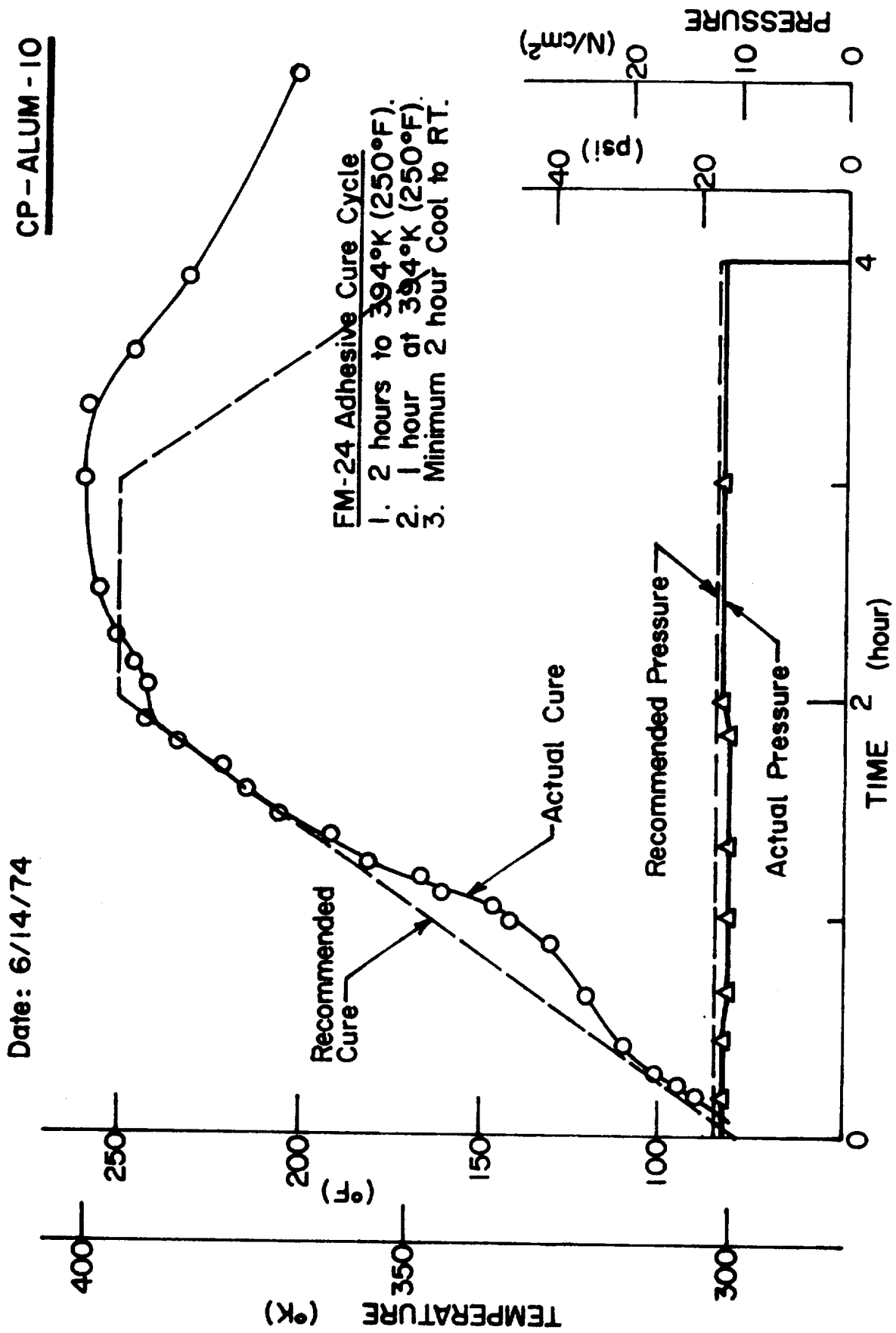


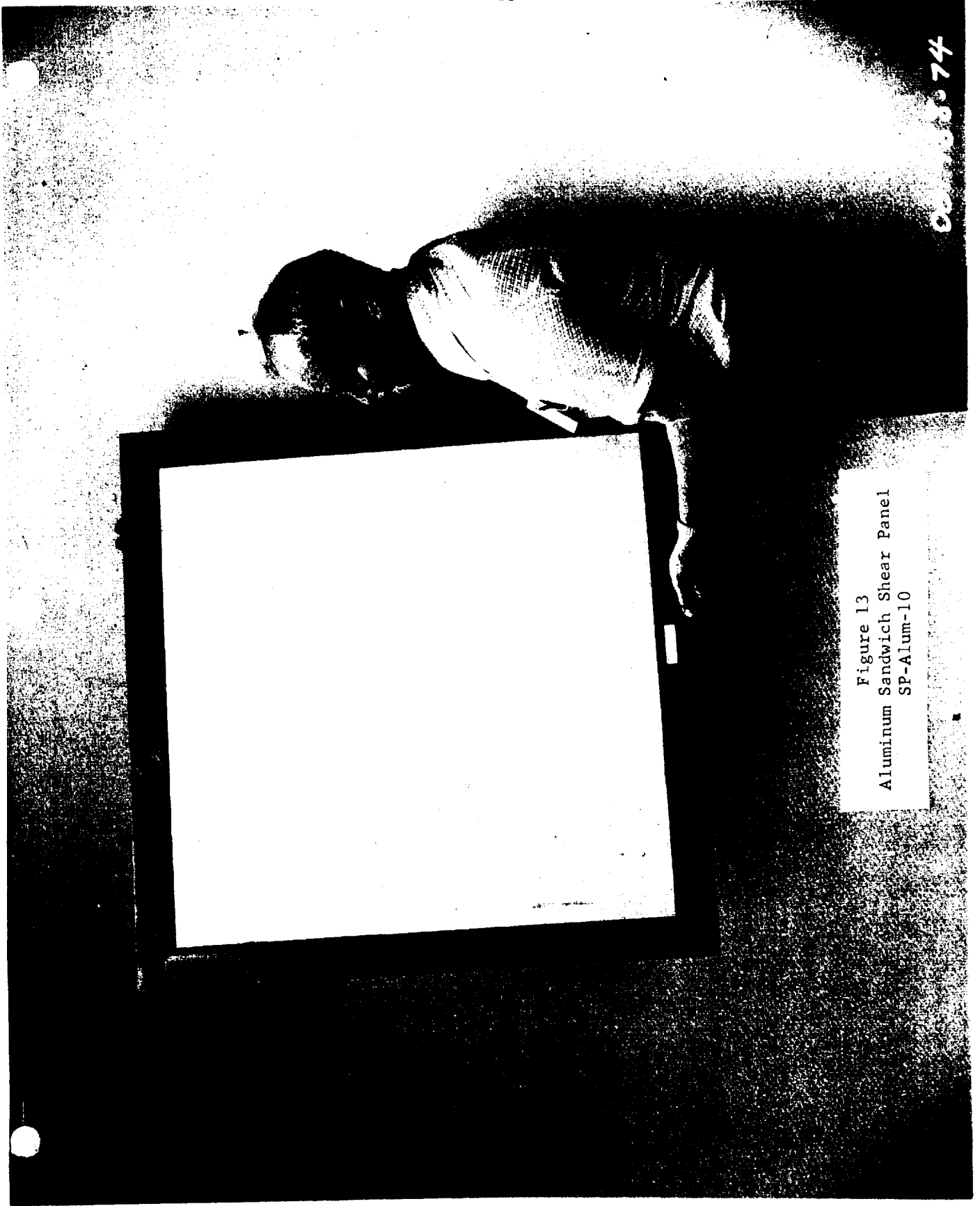
Figure 11
Modified FM-24 Cure Cycle
for Aluminum Sandwich Panels



Figure 12
Aluminum Sandwich Compression Panel
CP-Alum-10

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Figure 13
Aluminum Sandwich Shear Panel
SP-Alum-10



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C. Aluminum Truss Configuration

The aluminum truss configuration shown in Figure 14 contains tubular aluminum truss members mechanically fastened at the joints using doubler plates and blind cherry rivet fasteners. The inner and outer surfaces of the truss are covered with thin (0.010 cm) fiberglass cloth sheets to provide meteoroid protection. The flanges and webs of the truss horizontal and diagonal members are chemically milled to final dimensions. Detailed drawings for fabrication of three truss sections were provided in the previous quarterly report. The three components are a development panel (DP-ALUM-Truss), a compression test panel (CP-Alum-Truss) and a shear test panel (SP-Alum-Truss). Basic truss components are vertical stringers, selectively chem milled horizontal frames and diagonal stiffeners, joint doubler plates and blind cherrylock rivets.

Detail Fabrication - The aluminum doubler plates were made from 0.127 cm (0.050 in.) thick 2014-T6 aluminum alloy. Doublers were laid out by hand, cut and filed to size. Then one of each type was used as a drill template. Pilot holes (.040" diameter) were drilled into the template. The remaining doublers were stacked with the template on top, clamped and drilled.

All tubular details were initially cut one-quarter inch oversize. The vertical stringer tubes were simply trimmed at the ends to final size. The horizontal and diagonal members which required chem-milling were given a flash etch in an alkaline solution, water rinsed, submersed in an iridite solution for 10 minutes, water rinsed and wiped dry. Each tube was then plugged at one end with a silicone rubber plug that was expanded, once inside the tube, by compressing with two wing nuts on threaded rod. A silicone rubber plug with a stainless steel vent tube sealed the other end and was held in place with lead tape. The sealed tubes were individually dipped into a commercially available maskant solution (organoceram) that was thinned with xylene. Depending upon the thickness of the maskant, two or three coats produced a fully covered tube. Using a template, maskant on the sides to be chem-milled the deepest was cut away. A four tube assembly was mounted in a stainless steel fixture as shown in Figure 15, prior to chem milling. The tubes and fixture were immersed into the alkaline solution at 358°K (185°F) and two sides of each tube chem milled. The chem milling rate was approximately 0.0013"/minute (0.0005"/minute). The vent tubes provided an escape route for the hot, expanding air inside the tubes. Thickness was checked periodically during chem milling, and when a thickness was reached equalling the difference in thickness between the two sides, the maskant on the final two sides was cut away. Chem milling proceeded on all four sides until the desired chem mill depth was reached. The chem milled details showed a smooth fillet from the chem milled area into the original surface. A shallow, rounded ridge ran length-wise at the tube corners separating the sides of the tube.

The chem milled tube details were then cut to final size as shown in Figure 16. All tubes details were aged to the -T81 condition at $433^{\circ}\text{K} \pm 5^{\circ}\text{K}$ ($320^{\circ}\text{F} \pm 10^{\circ}\text{F}$) for 18 hours.

Panel Assembly - The panel assembly tool was simply a modified mill cutting table which provided a flat surface and a means of securing detail parts prior to attachment. The doublers at the ends of each long vertical stringer member were first attached using a weld bond technique. This process involved spot welding through a thin layer of adhesive to produce a high strength lightweight joint. Each end of the tube was first abrasively cleaned and wiped with a solvent, on both the inside and outside. The adhesive (Hysol ADX-347) was applied to the outside of each tube on both sides. Both doublers for that joint were positioned and clamped in place. The spot welding was then performed as shown in Figure 17 on both doublers at once using a copper bar machined to fit the inside of the tube. The vertical members, stringers, were next aligned in the fixture and clamped in place. The four horizontal end details were aligned with the doublers on the stringers, clamped in place and riveted. The remaining details were positioned, clamped in place and riveted. To insure proper fit-up between tube details, doublers and rivets, each end of each tube was marked in pencil with a centerline and two parallel guidelines 0.508 cm (0.200") from the side of the tube. Once the tube was positioned properly and clamped, the lines could be seen through the pilot holes in the doubler. The doubler was moved so that the middle pilot hole was centered on the center line and the outer two holes were between the two parallel side lines. In this way the as-fastened rivet did not extend onto the tube corner radius. Once the doublers were fitted to a joint, pilot holes were drilled through the doubler into the tube detail part. When enough holes were drilled to secure the doubler in place, "clicos" clamps were inserted and the remaining pilot holes drilled. The next step was to drill full size holes 0.510 cm (0.201") in diameter, insert large "clicos", removing the smaller ones, and finish drilling all holes (Figure 18). The doubler was then removed and all holes finished to size using a 0.520 cm (0.205") reamer. The holes were deburred and the surfaces cleaned. The doubler was repositioned using the large "clicos" and the rivets attached. The panel was then taken out of the fixture and turned to rivet the opposite side. The panel was shimmed in the fixture as shown in Figure 19 so that the rivet heads did not touch the assembly table. The same procedure was used to finish the second side. The completed panel was wiped with a solvent and the adhesive cured in an oven at 250°F for one hour. The compression test panel without fiberglass meteoroid protection layers is shown in Figure 20.

Fiberglass Meteoroid Protection - The completed aluminum truss was covered on both front and back surfaces as shown in Figure 21 with 0.010 cm (0.004 in.) thick fiberglass cloth for meteoroid protection. The cloth layers consist of precured single plies of style 120 glass cloth. The translucent layer of cured cloth was positioned on the truss and held with tape while holes were cut to accommodate the doubler plates. The trimmed layer was then removed, the truss members cleaned, adhesive added to the truss component surfaces and the cloth layer repositioned for bonding.

The average measured weight of the three truss panels, without special end attachment provisions, was 3.07 kg/in^2 (0.629 lb/ft^2).

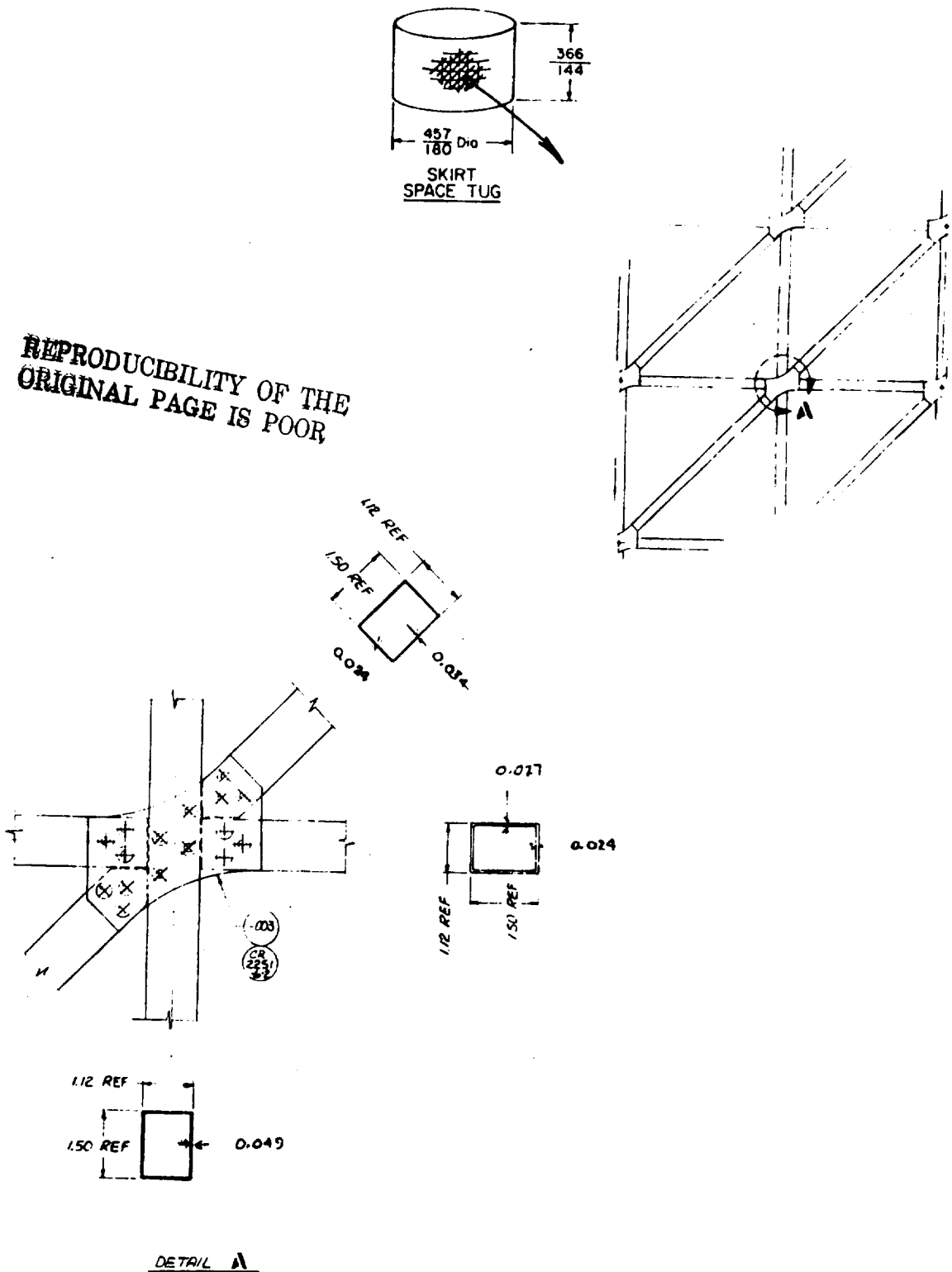


Figure 14. Aluminum Truss Configuration

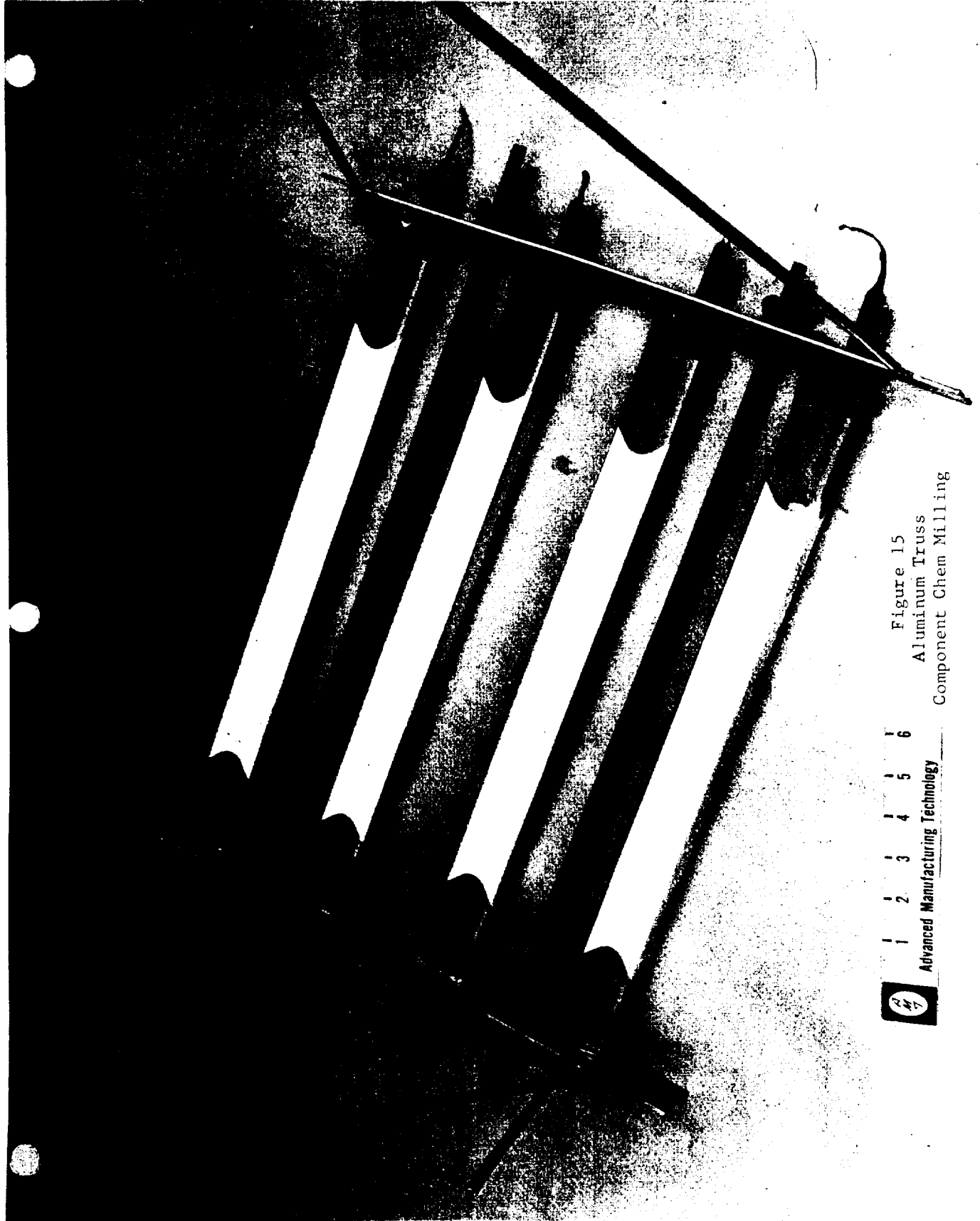


Figure 15
Aluminum Truss
Component Chem Milling



Figure 16
Aluminum Truss
Members

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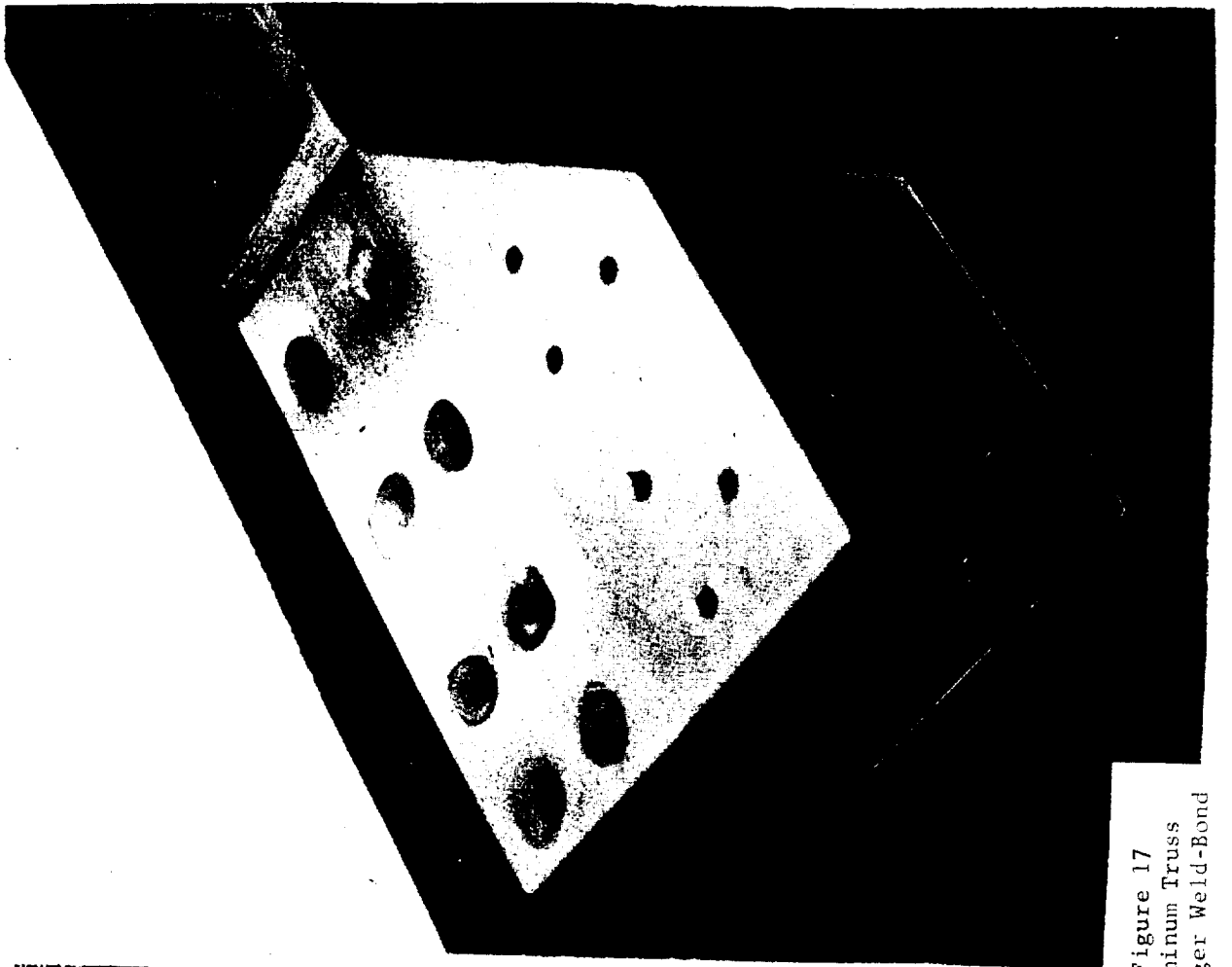


Figure 17
Aluminum Truss
Stringer Weld-Bond

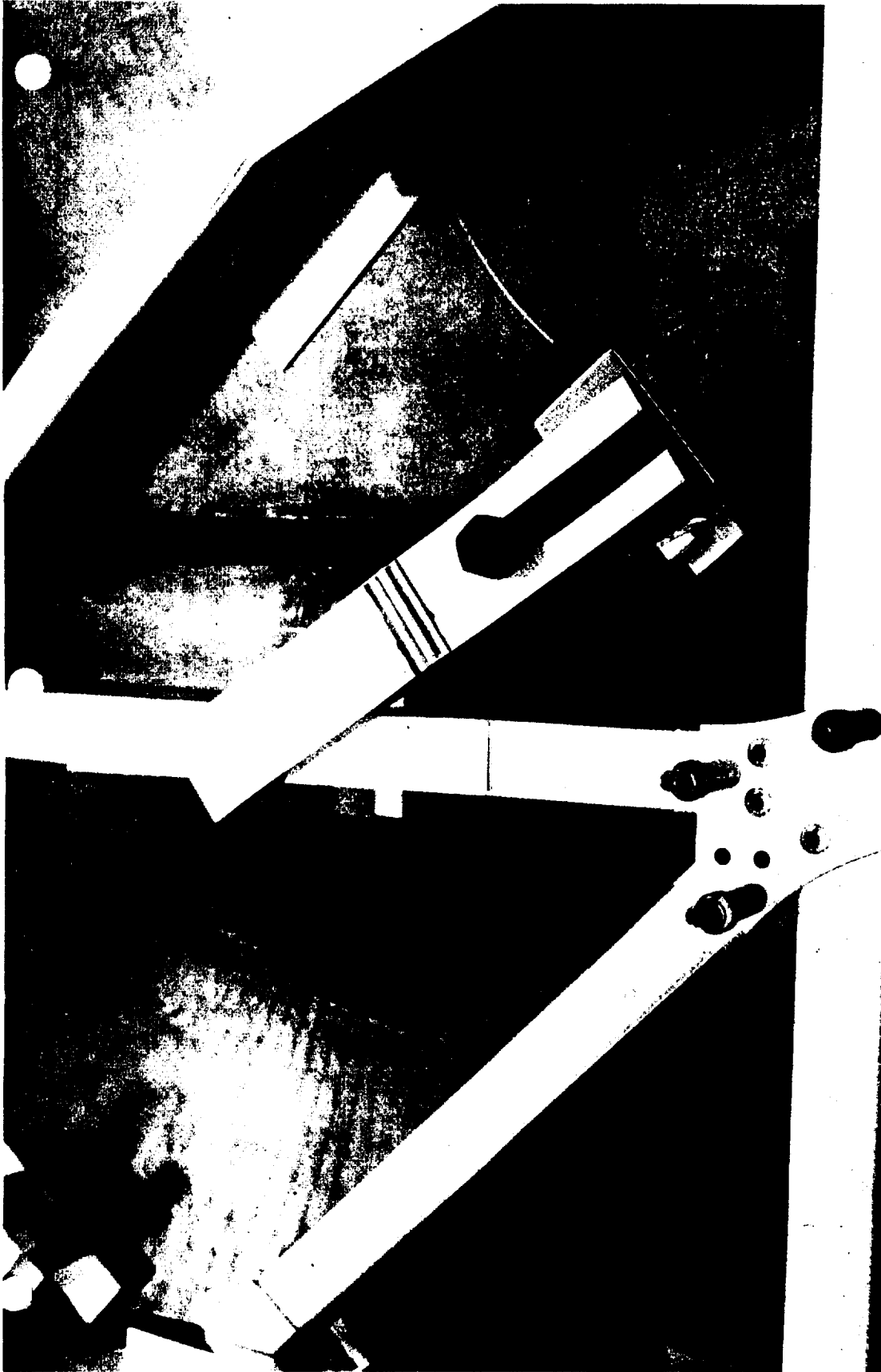


Figure 18
Aluminum Truss
Joint Assembly

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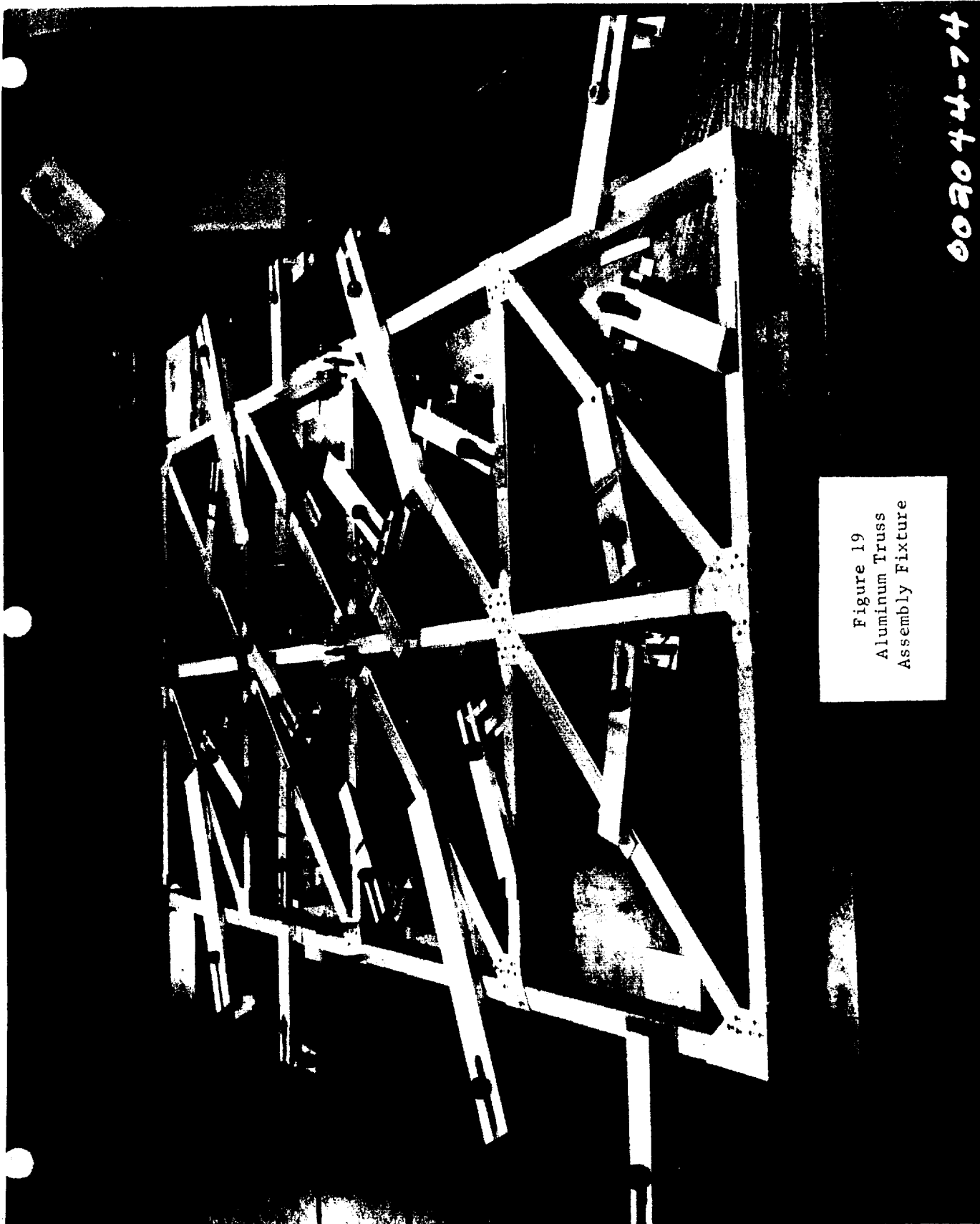


Figure 19
Aluminum Truss
Assembly Fixture

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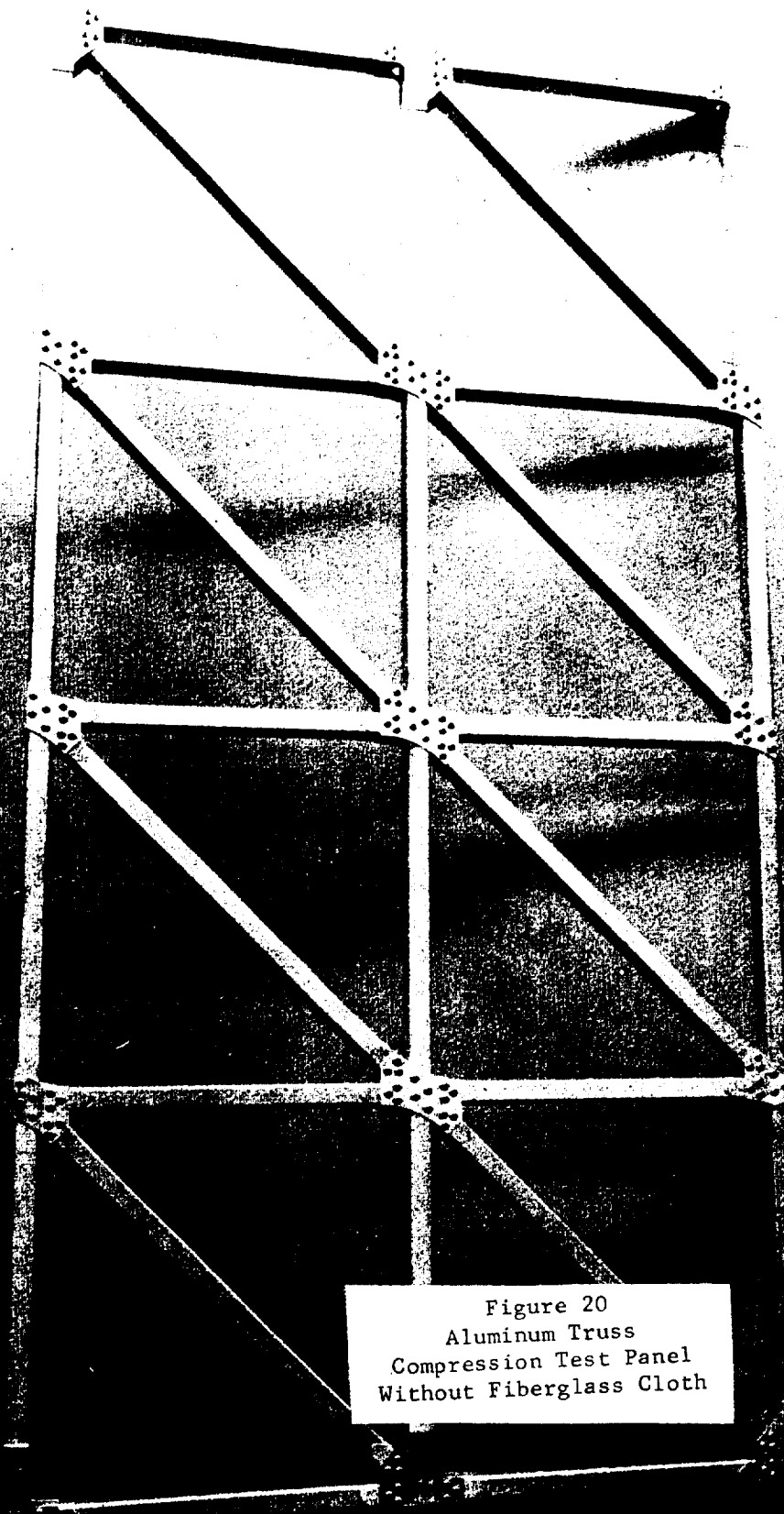


Figure 20
Aluminum Truss
Compression Test Panel
Without Fiberglass Cloth

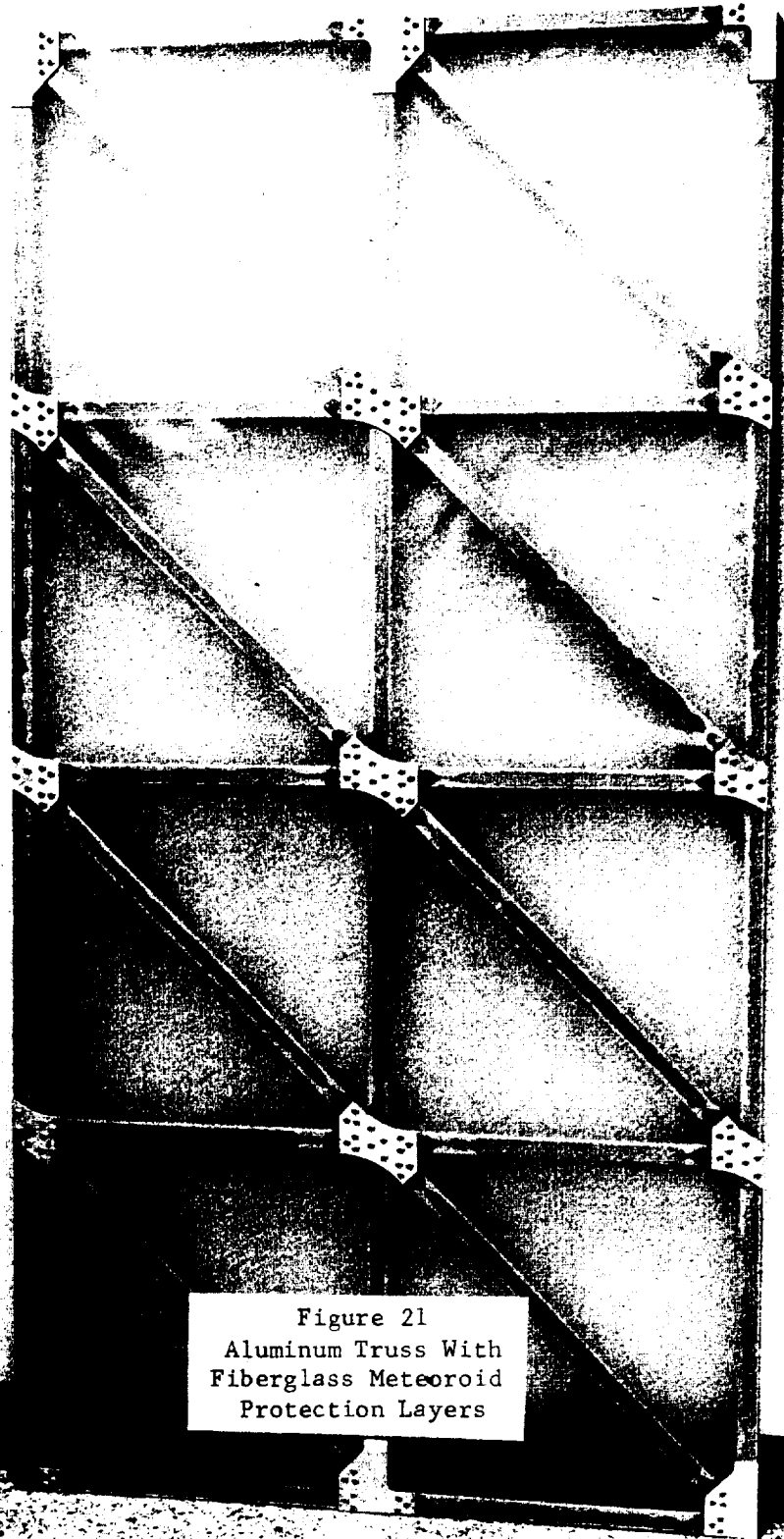


Figure 21
Aluminum Truss With
Fiberglass Meteoroid
Protection Layers

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D. Structural Test

The design and analysis study conducted during Phase I was concerned with a 3.66 m (144 inch) tall by 4.51 m (180 inch) diameter cylindrical shell structure subjected to combined loading of 1225.8 N/cm (700 lb/in) axial compression and 245.2 N/cm (140 lb/in) torsion. Preliminary evaluation of candidate concepts was aided by structural tests of small development panels. Three of the concepts were selected for further evaluation during Phase II. The development test panels for each of the three selected concepts consists of a flat 1.83 m by 0.92 m (6 ft by 3 ft) compression panel and a flat 0.92 m by 0.92 m (3 ft by 3 ft) shear panel. The compression panels are to be supported along all four edges and subjected to uniform axial compressive loading until failure. The shear panels are to be loaded to failure in pure shear, using an appropriate test support fixture. Unfortunately, the critical failure mode of the lightweight, large diameter cylinders designed during Phase I is overall instability at the combined design ultimate loading. The overall buckling characteristics of the flat test panels are not related to larger cylindrical shell buckling behavior and, therefore, must be investigated prior to structural test.

Compression Test Panel Overall Buckling - The buckling load of a rectangular sandwich plate with isotropic faceskins under uniaxial compression can be predicted from the expression.

$$P = k \frac{\pi^2 D}{b^2} \quad 1)$$

where b is the panel width and D is the bending stiffness per unit run calculated from

$$D = \frac{EI}{(1-\nu^2)} \quad 2)$$

where E is Young's modulus, I is moment of inertia and ν is Poisson's ratio. The buckling coefficient, k , depends on the boundary support, panel geometry and sandwich core shear stiffness. The shear stiffness is defined to be

$$s = \frac{b^2 S}{\pi^2 D} \quad 3)$$

where S is the transverse shear stiffness of the sandwich plate.

To determine the effect of core shear stiffness on panel buckling, the shear stiffness of the core in the warp or weak direction was used to calculate S from

$$S = G_w \left(\frac{c + f}{c} \right)^2 \quad 4)$$

where G_w is the core shear modulus, c is the core depth and f is the face-skin thickness. The resulting calculated value of s for the aluminum

sandwich compression panel was such that the buckling load reduction due to core shear stiffness was less than five (5) percent and was, therefore, neglected. The buckling coefficient, k , for the aluminum sandwich compression panel with $a/b = 2$, is dependent on the side boundary conditions of the plate. Values of k for several boundary conditions can be found in Reference 1. The structural test fixture will provide neither perfectly fixed nor simply supported boundary condition, therefore, calculations were made for these extremes with test values expected to fall between them if the designs prove to be buckling, rather than strength, critical. The value of k for simple support boundary conditions, $a/b = 2$ and $l/s \approx 0$ is given in Figure 4.2 of reference 1 to be 4.0. This results in a calculated panel buckling load from eqn. 1 of 1333 N/cm (762 lb/in).

If it is assumed that the panel boundaries are perfectly fixed, the k value from Figure 4.11 of reference 1 is 7.0 and the corresponding buckling load is 2325 N/cm (1330 lb/in).

Similar calculations can be made for the orthotropic graphite/epoxy panel. Assuming negligible core shear effect, equation 1 is again applicable, however, the bending stiffness D is replaced by

$$\bar{D} = \sqrt{D_x D_y} \quad 5)$$

where

$$D_x = \frac{E_x I_x}{(1 - \nu_{xy} \nu_{yx})} \quad \& \quad D_y = \frac{E_y I_y}{(1 - \nu_{xy} \nu_{yx})} \quad 6)$$

which takes into account the orthotropic nature of the graphite/epoxy faceskins.

The case of simply supported rectangular plates with $D_x \neq D_y$, loaded in uniaxial compression, is discussed in section 5.3 of Reference 1 and curves for k are shown in Figure 5.7 of that reference. The buckling coefficient for the graphite/epoxy sandwich test panel, is 3.1 which yields a critical buckling load for simple support boundary conditions of 992 N/cm (566 lb/in). The case of fixed boundary conditions with $D_x \neq D_y$, is presented in reference 2. The value of k for this condition is 6.0 which yields a critical buckling load of 1915 N/cm (1096 lb/in).

The critical overall buckling behavior of the aluminum truss can also be determined by calculating smeared D_x and D_y bending stiffnesses and treating the equivalent orthotropic panel. The critical buckling load, assuming simple support edge condition, calculated from eqn. 1 is 1505 N/cm (858 lb/in) and for the fixed edge support is 2900 N/cm

Reference 1 - Sandwich Construction, Plantems F.F., John Wiley & Sons, 1966.

Reference 2 - C.C. Chang, I.K. Ebcioğlu and C.H. Hsight, "General Stability Analysis of Orthotropic Sandwich Panels for Four Different Boundary Conditions, ZAMM, 42, 9 Sept. 1962, 373-389.

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(1660 lb/in).

Shear Panel Overall Buckling - The buckling load of sandwich panels loaded in pure shear can be predicted from

$$P = k \frac{\pi^2 \bar{D}}{b^2}$$

7)

where \bar{D} and b are as defined for compression buckling and k , the buckling coefficient, is determined from the boundary conditions, panel geometry and core shear stiffness. The appropriate values of k for simple support and fixed boundary conditions for the aluminum honeycomb shear panel taken from the NASA Design Structures Manual, Figure C2.1.5-14 are 9.5 and 15 respectively, if it is assumed that core shear stiffness is adequate. The calculated critical buckling loads are 3170 N/cm (1815 lb/in) for simple support boundary conditions and 5000 N/cm (2860 lb/in) for fixed boundary support.

Buckling coefficients for the graphite/epoxy panel are not readily available, however, it can be easily shown that the test panel is strength, rather than buckling, critical. If, for example the lowest bending stiffness, D_y is used rather than $\bar{D} = \sqrt{D_x D_y}$ and the isotropic buckling coefficients of 9.5 and 15 used again, the critical buckling loads are 1260 N/cm (721 lb/in) and 1990 N/cm (1139 lb/in) for simple and fixed support, respectively. Full development of panel strength would cause failure at approximately 1340 N/cm (766 lb/in). Since the D_x bending stiffness is five (5) times greater than the D_y value, it can be conservatively assumed that the true critical buckling load is much higher than the critical strength load.

Similarly, the overall buckling load for the aluminum truss shear panel can be shown to be significantly higher than the critical strength value by considering smeared D_x and D_y bend stiffnesses. If the lowest bending stiffness, D_y , is used rather than $\bar{D} = \sqrt{D_x D_y}$, the critical buckling loads are 3710 N/cm (2120 lb/in) and 5860 N/cm (3350 lb/in) for simple and fixed support respectively. These values are significantly higher than the expected critical strength value of 996 N/cm (570 lb/in).

Local Instability - Another possible mode of failure for each of the six (6) test panels is local instability. In the case of the sandwich panels this includes intercell buckling and face wrinkling and for the truss panels, local crippling of the tubular members.

1. Honeycomb Sandwich Face Wrinkling

The wrinkling phenomenon is a short wave faceskin buckling highly dependent on the transverse normal stiffness of the core. The critical faceskin load, P , can be calculated (Reference 1) from

$$P = 1.52 f (G_c E_{cz} E_f)^{1/3}$$

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where f is the faceskin thickness, G_c the core shear modulus, E_{cz} the core transverse normal stiffness and E_f the faceskin stiffness. The calculated critical faceskin wrinkling load for the aluminum sandwich panel is 8660 N/cm (4900 lb/in) or a faceskin stress of 168,500 N/cm² (245,000 psi). Similar calculations for the graphite/epoxy sandwich panels yield 16,080 N/cm (9180 lb/in) and 198,000 N/cm² (287,000 psi) if the faceskin stiffness E_f is taken to be the axial faceskin modulus, E_x , of the graphite/epoxy laminate.

2. Honeycomb Sandwich Intercell Buckling

The stress level at which a sandwich faceskin loaded in compression buckles locally within an individual hexagonal cell can be calculated (reference 1) from the expression

$$\sigma_{cr} = 3E \left(\frac{f}{d}\right)^2$$

where E is the faceskin modulus, f is the faceskin thickness and d is the cell size. The calculated values for the aluminum and graphite/epoxy sandwich panels are 132,000 N/cm² (192,000 psi) and 540,000 N/cm² (785,000 psi) respectively. The value for the graphite/epoxy panels was calculated using the faceskin axial Young's modulus E_x .

3. Truss Tube Local Crippling

The critical local crippling stress of thin walled rectangular tubing can be calculated from

$$\sigma_{cr} = k_h \frac{\pi^2 E}{12(1-\nu^2)} \left(\frac{t_w}{h_w}\right)^2$$

where E is Young's modulus, ν is Poisson's ratio, t_w is web thickness, h_w is web height and k_h is a coefficient dependent on stiffener geometry available from the NASA Structures Design Manual, Figure 4.2.2-5. The calculated critical stress values for the truss stringers, horizontals and diagonals are 33,100 N/cm² (48,100 psi), 10,280 N/cm² (14,900 psi) and 10,500 N/cm² (15,310 psi), respectively.

Material Strength and Stiffness - Mechanical properties of the materials used in fabricating the compression and shear test panels are listed in Table 1. The aluminum sandwich panel used 2014-T6 aluminum faceskins that were chemically milled from 0.102 cm (0.040 inches) down to 0.025 cm (+0.005 cm, -0.000 cm) (0.010 inches, -0.000 inches) (+0.002 inches, -0.000 inches). It has compression and shear ultimate values of 40,000 N/cm² (58,000 psi) and 26,850 N/cm² (39,000 psi) respectively. The rectangular aluminum tubing used in the truss structure was 2024 alloy that was received in the T3 hardened condition and heat treated to the T81 condition. This yielded tubing with ultimate compression and shear strength values of 39,250 N/cm² (57,000 psi) and 24,100 N/cm²

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(35,000 psi) respectively. The graphite/epoxy laminates used as faceskins on the sandwich panels consist of two layers of axial Type I/5208 material sandwiched between four symmetric layers of $\pm 45^\circ$ T-300/5208 material. A typical cured faceskin thickness was 0.041 cm (0.016 in) of which 0.023 cm (0.009 in) was axial material and 0.018 cm (0.007 in) was $\pm 45^\circ$ material. The axial strength and stiffness shown in Table 1 for this laminate was taken from Phase I small panel test results while other values are estimated from material properties of the constituents.

Test Results Summary

The calculated theoretical critical loads and the actual test loads for the six (6) panel structural tests are listed in Table 2. The honeycomb sandwich compression panel with aluminum faceskins shown in the test fixture in Figure 22 was critical in overall panel buckling. The panel was loaded to 1285 N/cm (735 lb/in) without failure. The test was terminated at that point since excessive center panel normal deflection indicated the onset of panel buckling. Terminating the test within the elastic strain range allows for possible future retest under different test conditions. The test information is sufficient to predict an overall buckling load, from a Southwell plot (Figure 23) of test data, of 1607 N/cm (918 lb/in). Similarly, the test of the sandwich compression panel with graphite/epoxy faceskins was terminated at an applied load of 1268 N/cm (725 lb/in) without failure. A Southwell plot of test data shown in Figure 24 was used to predict a buckling load of 1362 N/cm (779 lb/in). The critical failure mode of the aluminum truss compression panel was local instability of the stringer segments. Simultaneous catastrophic failure of all three stringer sections (Figure 24) occurred at a test load of 1073 N/cm (613 lb/in). Local buckling failure occurred at a stress level approximately 13 percent lower than was predicted.

The aluminum sandwich shear panel failed (Figure 26) at 1594 N/cm (911 lb/in) with faceskin principal strains all well beyond the elastic yield strain. Similarly, the graphite/epoxy shear panel failure occurred at material strain levels indicative of full development of material strength. The failure did not initiate at the faceskin repair patch as can be seen in Figure 27. The overall panel shear load at failure was 1520 N/cm (869 lb/in). The critical failure mode of the aluminum truss shear panel was local instability of the diagonal truss members. Initial buckling occurred at an effective shear load of 245 N/cm (140 lb/in), however, initial buckling did not cause catastrophic failure due to the low stress level at which it occurred. The panel failed catastrophically (Figure 28) at 542 N/cm (310 lb/in). A full test report including strain gage and deflectometer data is currently being written.

In summary, both of the honeycomb sandwich concepts, aluminum and graphite/epoxy, use faceskins which are minimum gage as determined

by fabricability, handleability, available raw material size, quality assurance and damage sensitivity and each has adequate strength and stiffness. The aluminum truss concept requires only slight design modification to satisfy strength requirements, however, the redesigned truss would still have a smaller margin of safety than the sandwich panel concepts.

TABLE 1
MATERIAL PROPERTIES

PROPERTY	ALUMINUM SANDWICH FACESKINS,* 2014-T6	ALUMINUM TRUSS TUBES,* 2024-T81	GRAPHITE/EPOXY SANDWICH FACESKINS, TYPE I/T-300/5208
Compression Ultimate σ_{cy} , N/m ² , (psi)	58,000	57,000	54,000
Shear Ultimate τ_u , N/m ² , (psi)	39,000	35,000	24,000
Axial Young's Modulus E_x , N/m ² (psi)	10,500,000	10,500,000	15,850,000
Transverse Young's Modulus E_y , N/m ² (psi)	10,500,000	10,500,000	2,740,000
Shear Modulus G_{xy} , N/m ² (psi)	4,000,000	4,000,000	2,500,000
Poisson's Ratio ν_{xy}	0.33	0.33	0.25

*Values taken from MIL- Handbook 58, Sept. 1971.

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Table 2
Theoretical and Experimental Panel Loads

TEST SPECIMEN	THEORETICAL CRITICAL LOADS, (LB/IN)					STRUCTURAL TEST LOAD, (LB/IN)	COMMENT
	CRITICAL OVERALL BUCKLING LOADS		CRITICAL LOCAL BUCKLING LOAD	CRITICAL STRENGTH			
	FIXED SUPPORT	SIMPLE SUPPORT					
Aluminum Sandwich Compression Panel	1330	762	3,840	1275	735	No failure, 918 lb/in predicted southwell buckling load	
Graphite/Epoxy Sandwich Compression Panel	1096	566	9,180	1730	725	No failure, 779 lb/in predicted southwell buckling load	
Aluminum Truss Compression Panel	1660	858	708	840	613	Stringer local buckling failure	
Aluminum Sandwich	2860	1815	3,840	858	911	Material strength failure above yield stress	
Graphite/Epoxy Sandwich Shear Panel	2740*	1740*	6,700	766	869	Material strength failure	
Aluminum Truss Shear Panel	3359*	2120*	159	570	310	Diagonal member local buckling at 140 lb/in, failure at 310 lb/in	

*Estimated using $\bar{D} = \sqrt{D-D}$ and $K = 9.5$. $K = 15.0$

*Estimated using $\bar{D} = \sqrt{D_x D_y}$ and $K_s = 9.5$, $K_f = 15.0$

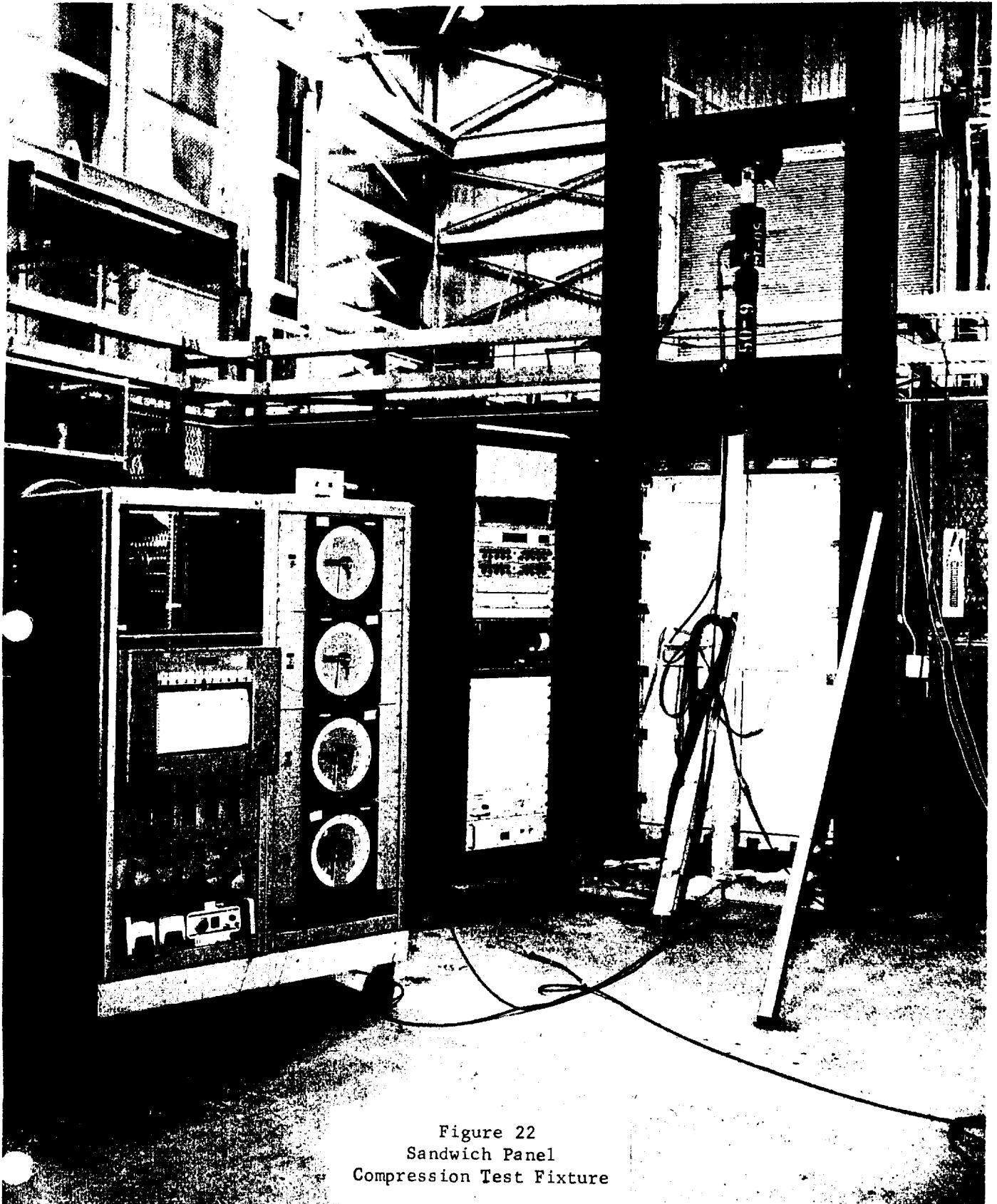


Figure 22
Sandwich Panel
Compression Test Fixture

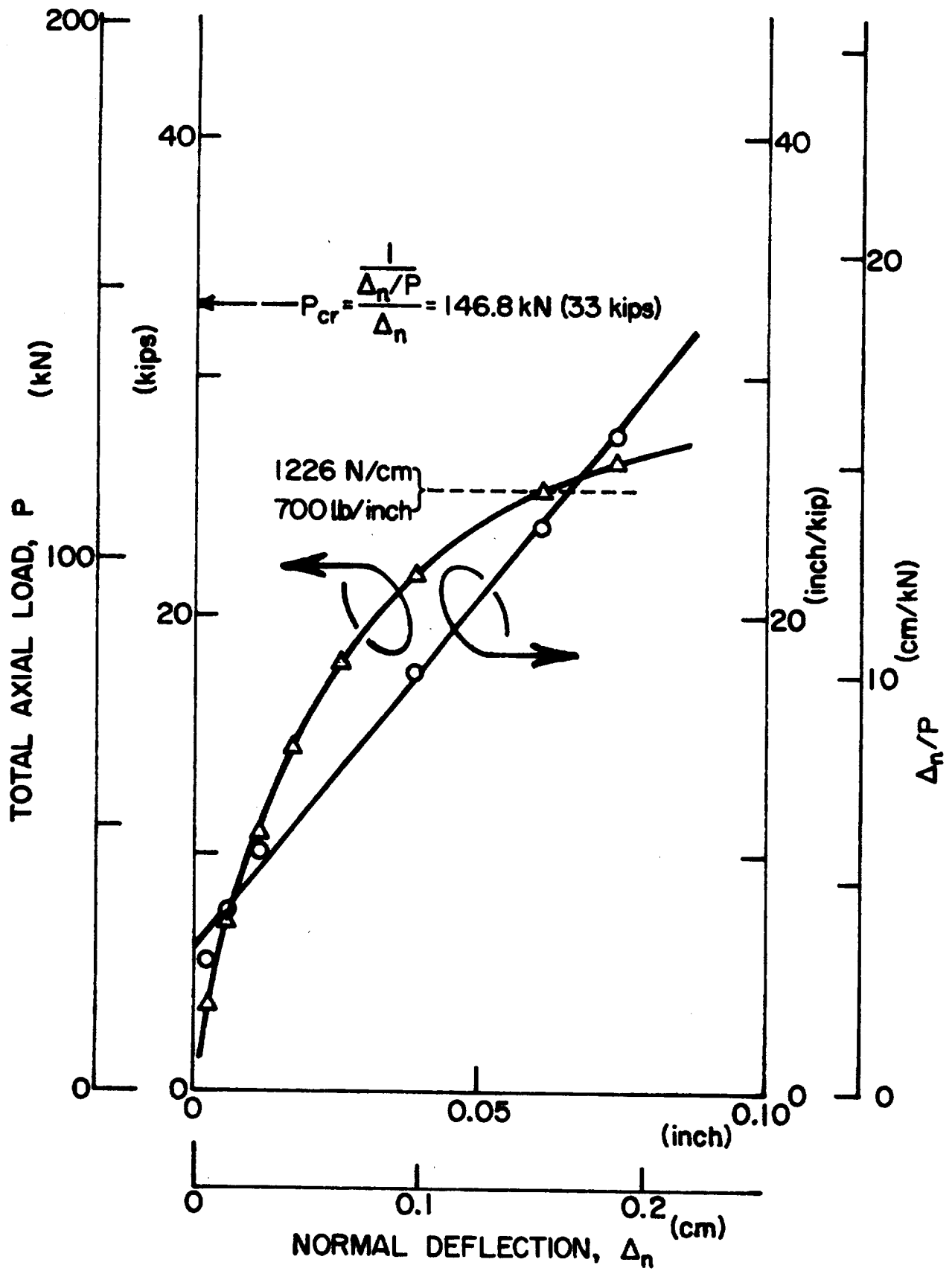


Figure 23
Southwell Plot
Aluminum Panel Compression Test

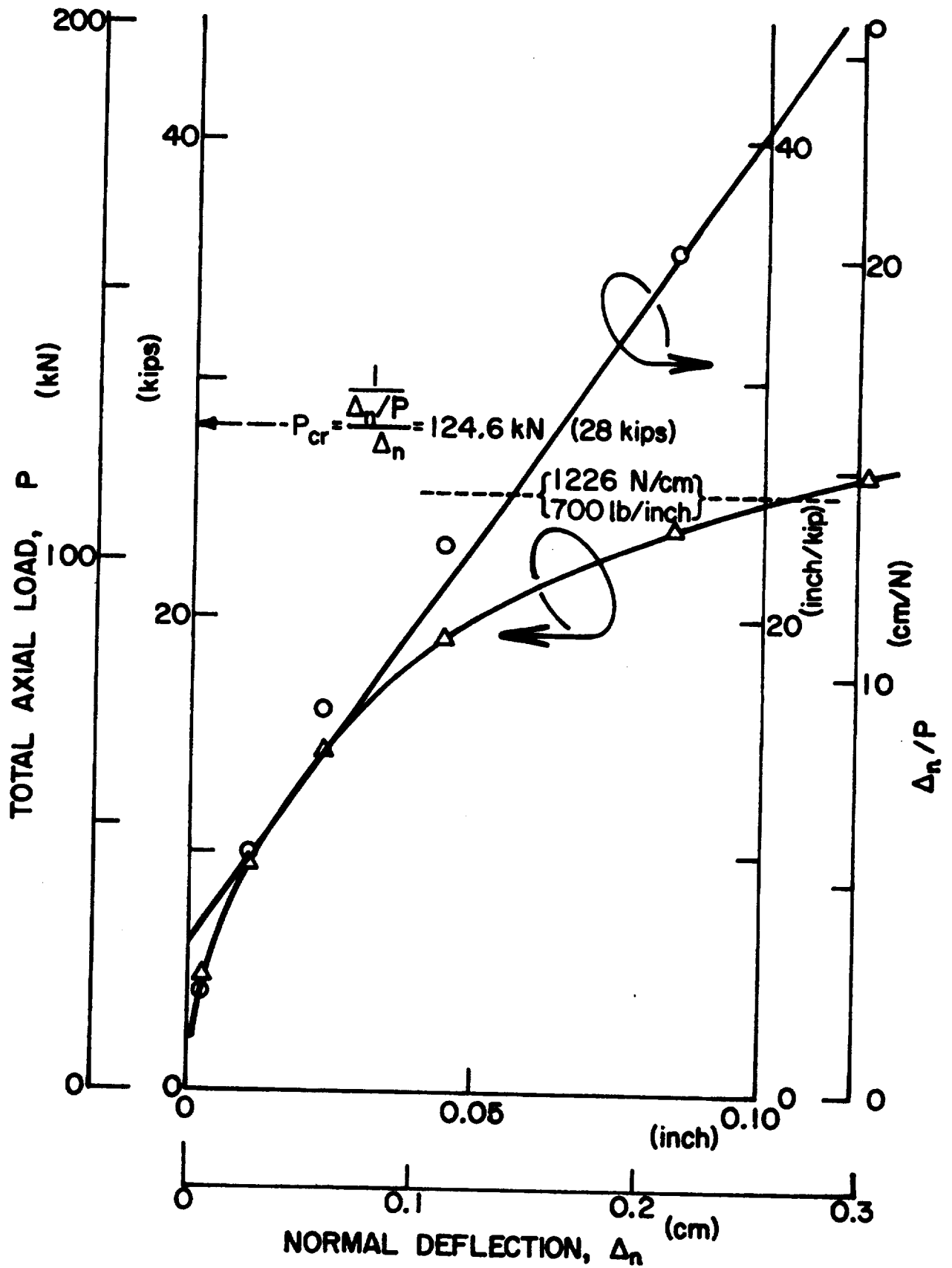


Figure 24
Southwell Plot
Graphite/Epoxy Panel
Compression Test

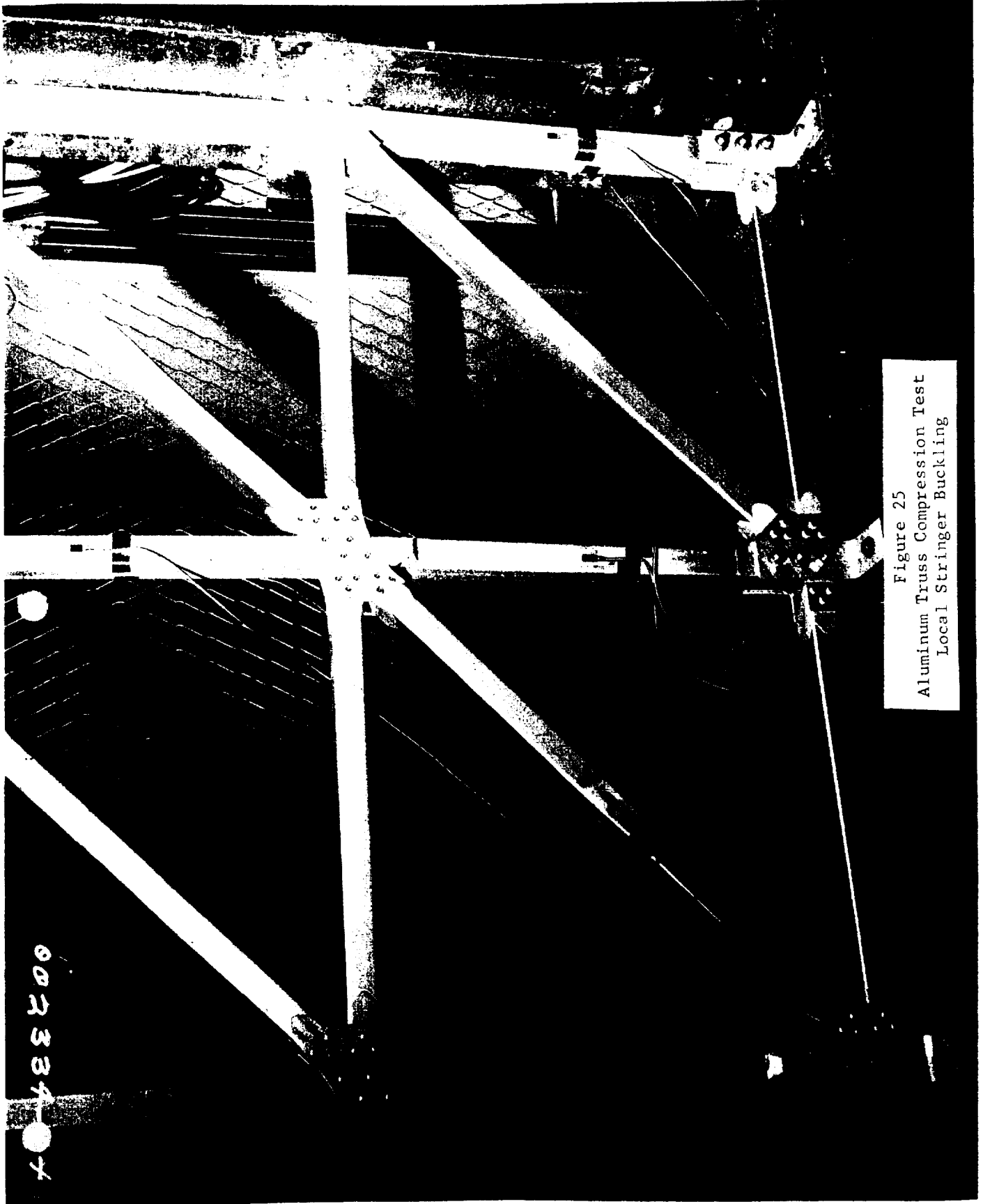


Figure 25
Aluminum Truss Compression Test
Local Stringer Buckling

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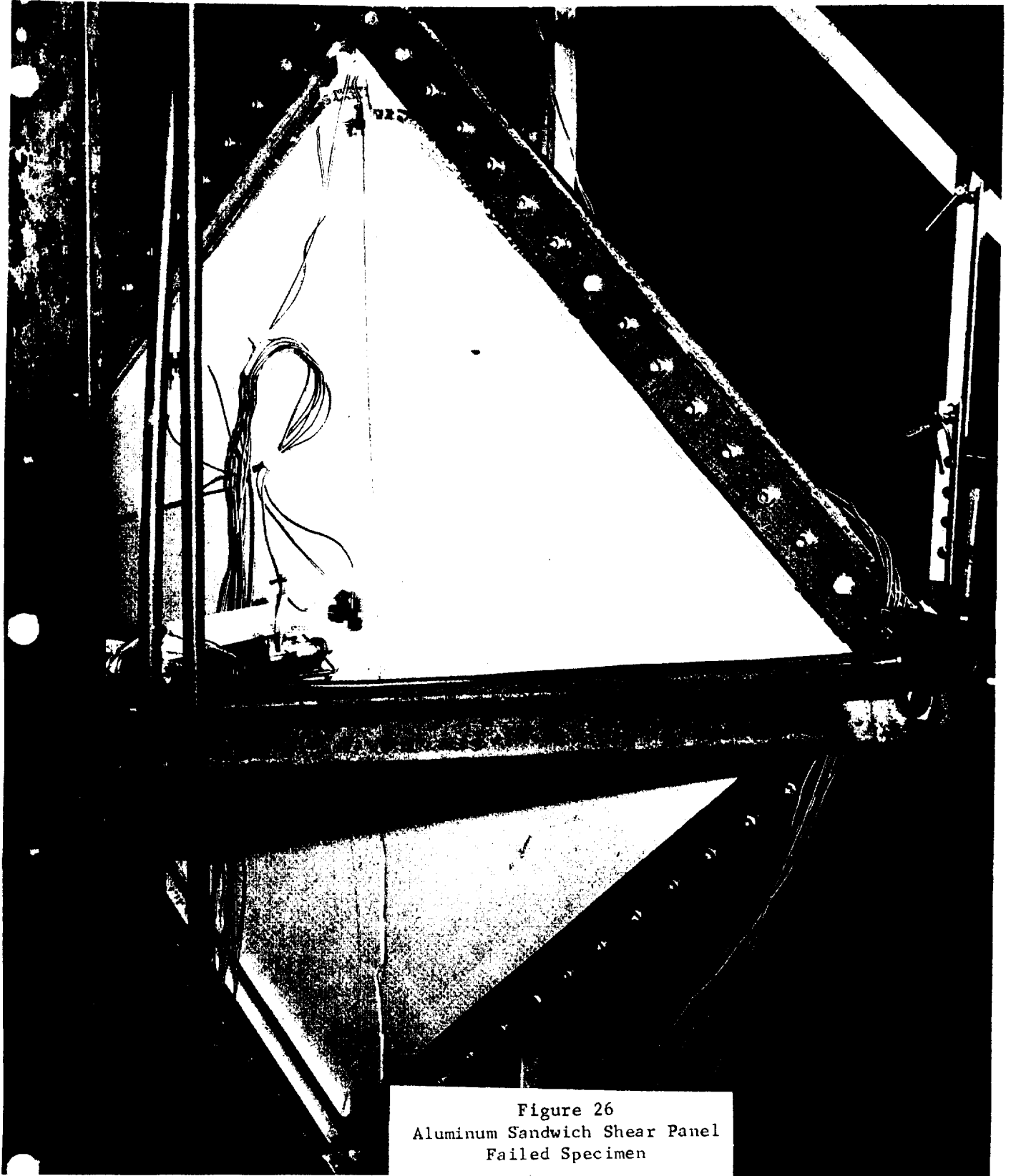
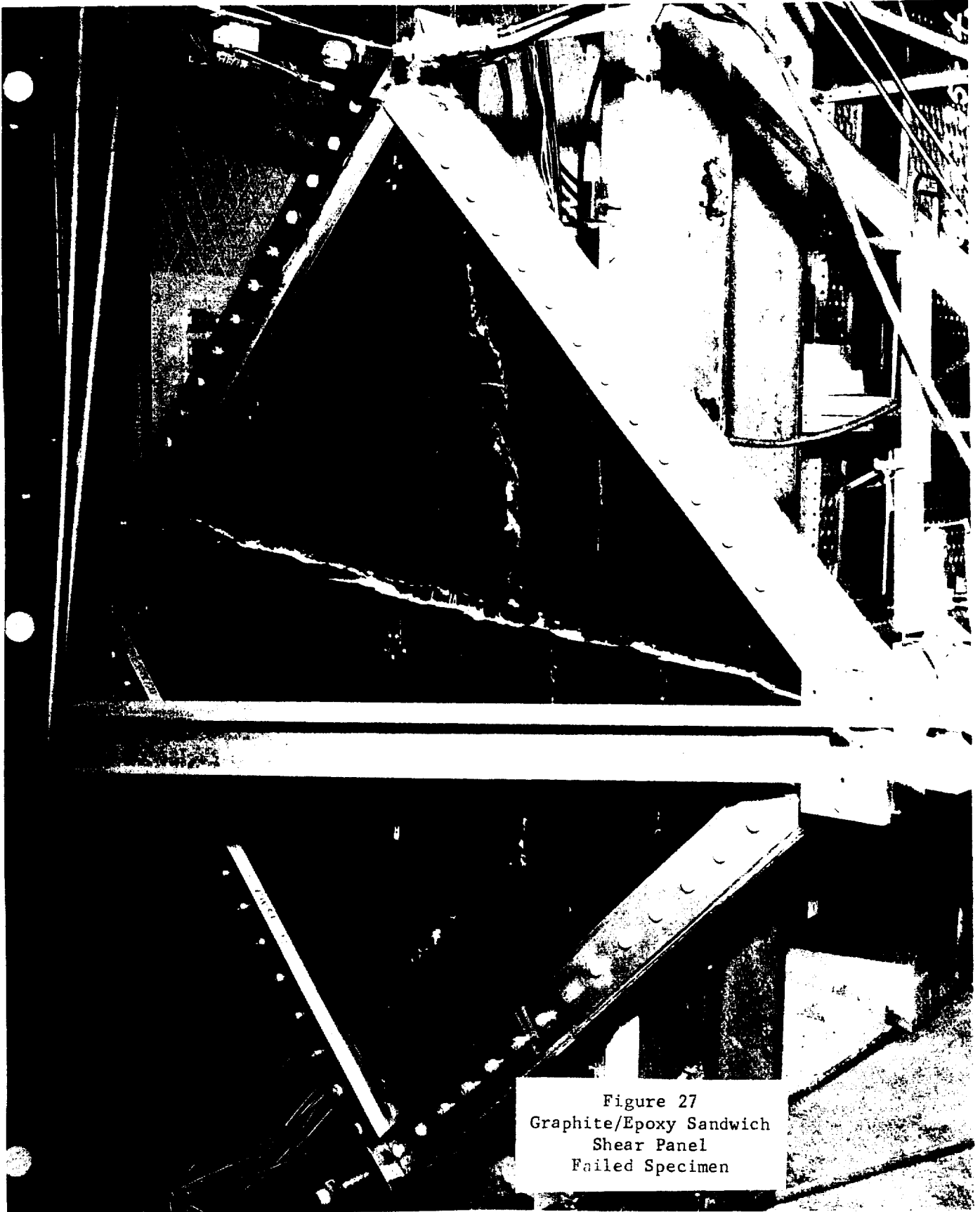


Figure 26
Aluminum Sandwich Shear Panel
Failed Specimen



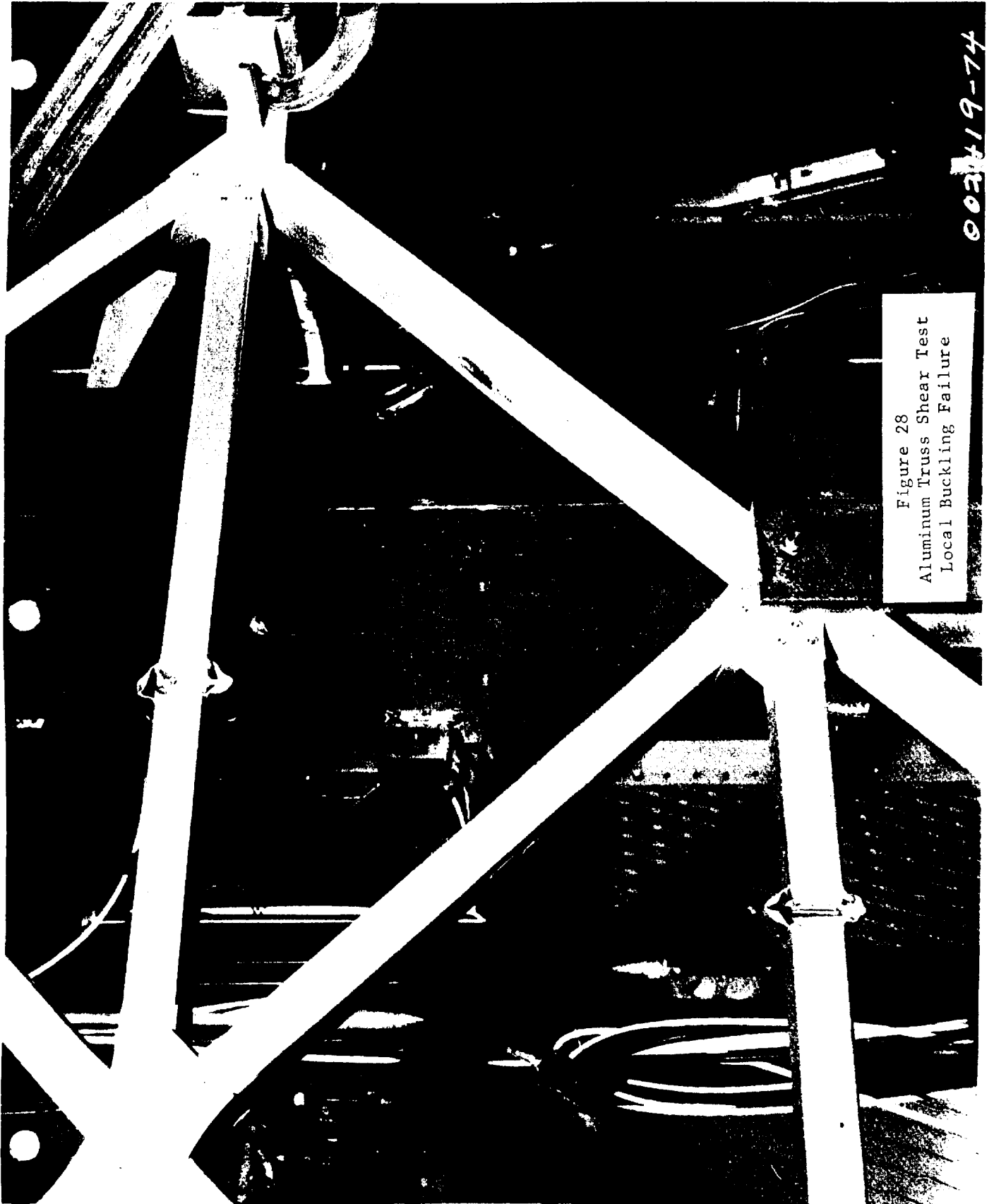


Figure 28
Aluminum Truss Shear Test
Local Buckling Failure

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E. Quality NDE

An aluminum honeycomb sandwich panel, identical in basic construction to the Phase II test panels, was fabricated with a wide variety of included defects to determine the effectiveness of two nondestructive evaluation (NDE) methods, ultrasonic and radiographic inspection. This control panel was divided into four quadrants as shown in Figure 29, each containing different types of defects. In addition, a honeycomb core splice was made down the center of the panel, with the core splice adhesive purposely contain defects as shown in Figure 30. Also shown in Figure 30 is the core damage introduced to quadrant B and two strands of style 120 fiberglass cloth, one coated for use as a release cloth and the other untreated bleed cloth. Two additional strands of these same materials are shown in Figure 31 located on top of the FM-24 adhesive film. Also shown are the gaps and overlaps in the FM-24 film introduced into quadrant A. The side of the aluminum faceskin that was placed against the adhesive film is shown in Figure 32 with the grease spot, faceskin scratch and faceskin dent visible. In addition, a single drop of water was included in one of the core cells of quadrant D. This control panel was assembled and subjected to a cure cycle identical to that of the aluminum sandwich panels for Phase II structural testing.

The completed control panel was ultrasonically and radiographically inspected by MMC quality assurance personnel without prior knowledge of the location, type or extent of included defects. Ultrasonic C scans of the four quadrants are shown in Figures 33 thru 36. The C scan of quadrant A, shown in Figure 33 revealed all of the FM-24 splice defects and also shows the core splice. The C scan of quadrant B, shown in Figure 34, revealed anomalies at the locations of local core cell wall surface crushing and buckling but did not reveal core cell wall wrinkling through the depth of the core. Neither the faceskin dent or scratch were revealed on the C scan of quadrant C, shown in Figure 35. The faceskin dent was flattened out by the pressure applied during panel fabrication and therefore would not be expected to reveal a C scan anomaly. The grease spot introduced into quadrant D is readily detectable on the C scan shown in Figure 36. The location of the water drop in the core cell is also shown due to the apparent bond problem caused by the resultant steam during panel cure at 250°F. The location of the 0.25 inch gap in the core splice is also indicated on this C scan. The included fiberglass cloth strands with bond release coating are vaguely detectable; however, the untreated cloth strands are not detectable.

The control panel was next radiographically inspected to evaluate this NDE method. The only defects that were detectable were the core damage areas of quadrant B. An X-ray of this quadrant is shown in Figure 37. All four of the core defects indicated are readily detectable using appropriate x-ray viewing equipment.

In conclusion, the results of the NDE study are very encouraging. All of the included defects that would be expected to be of concern for maintaining structural integrity were detected by one or both of the inspection techniques.

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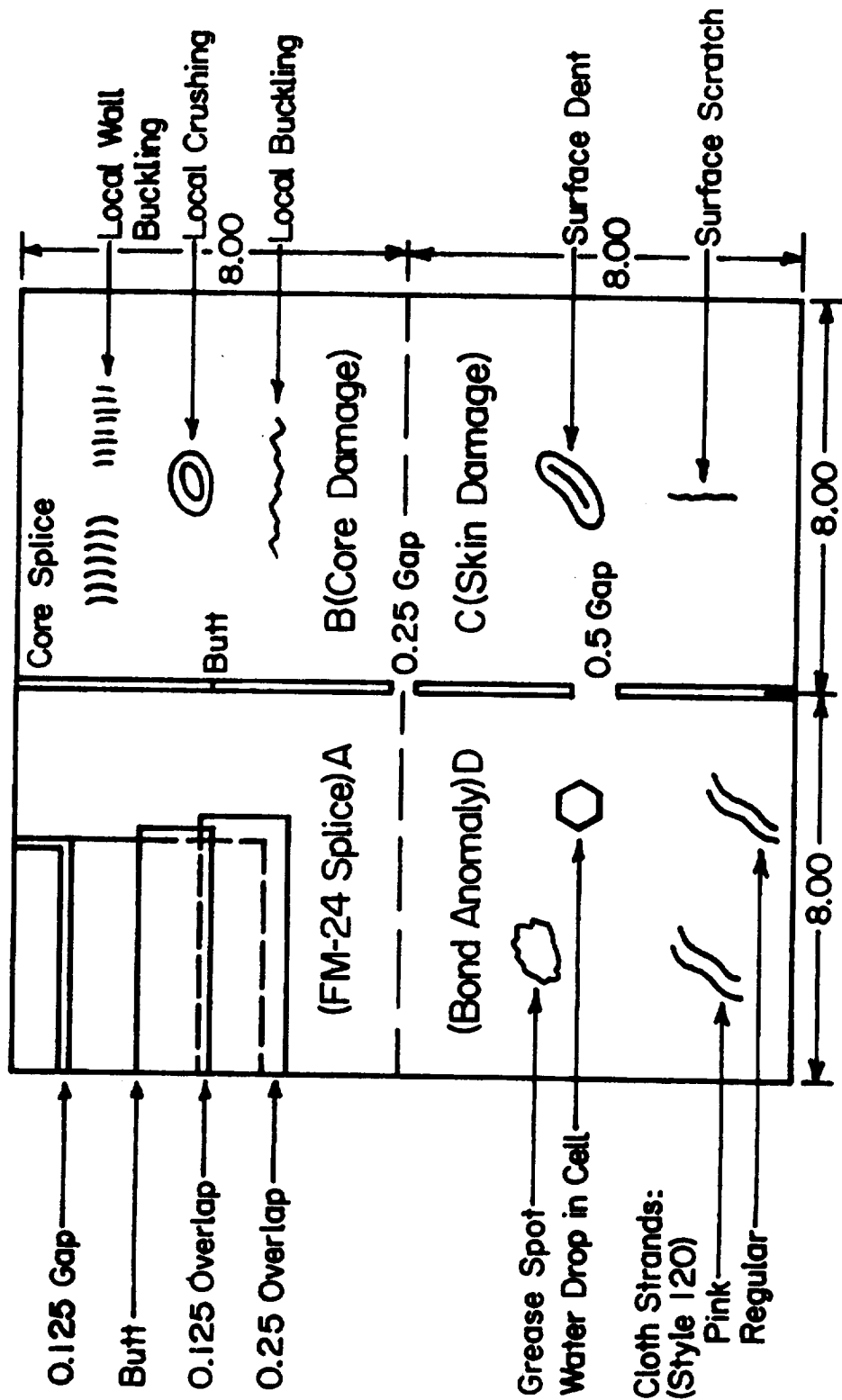


Figure 29

Aluminum Sandwich Panel
Quality NDE Standards Panel

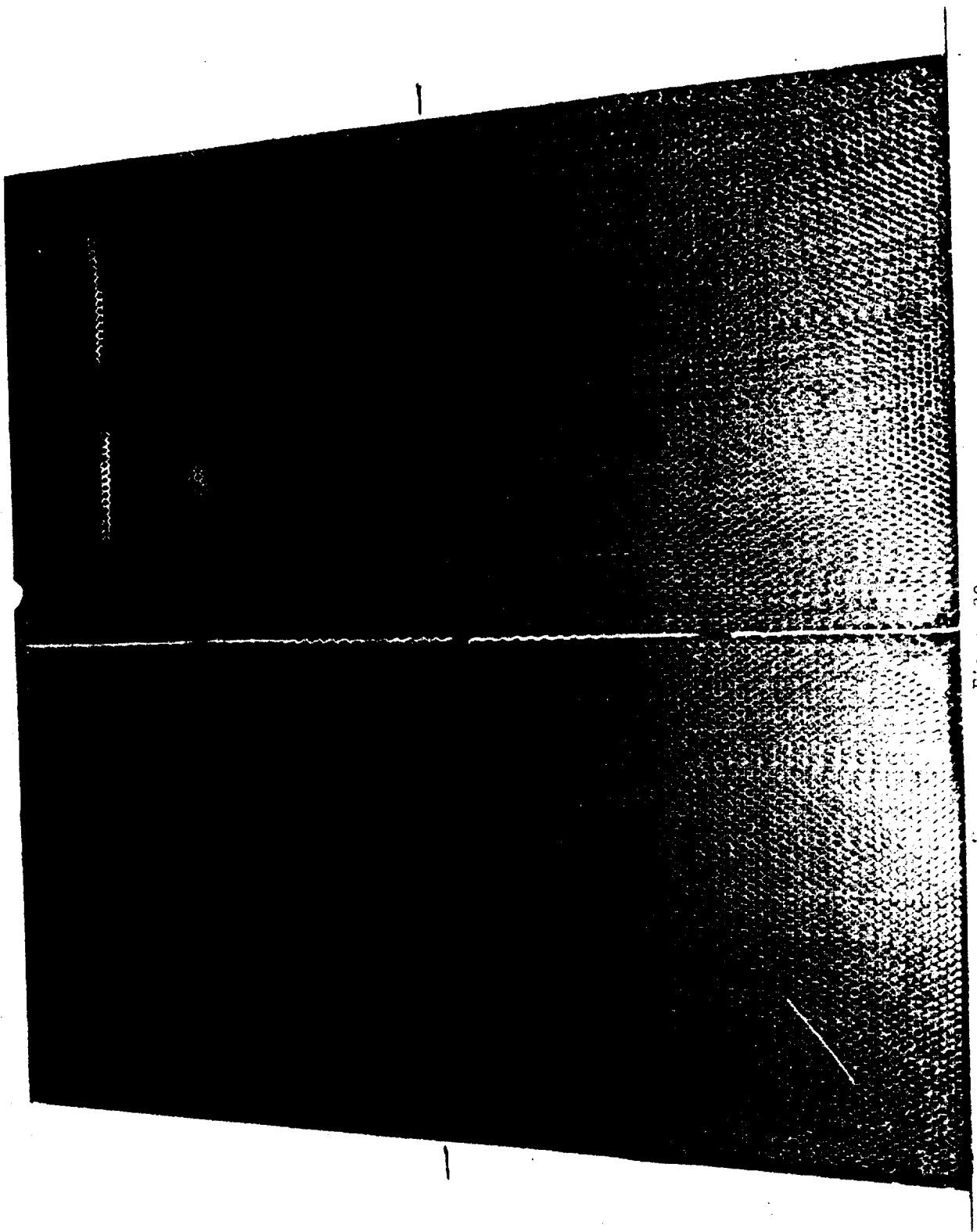


Figure 30

Quality NDE Standards Panel
Aluminum Core.

Figure 31
Quality NDE Standards Panel
FM-24 Adhesive Film

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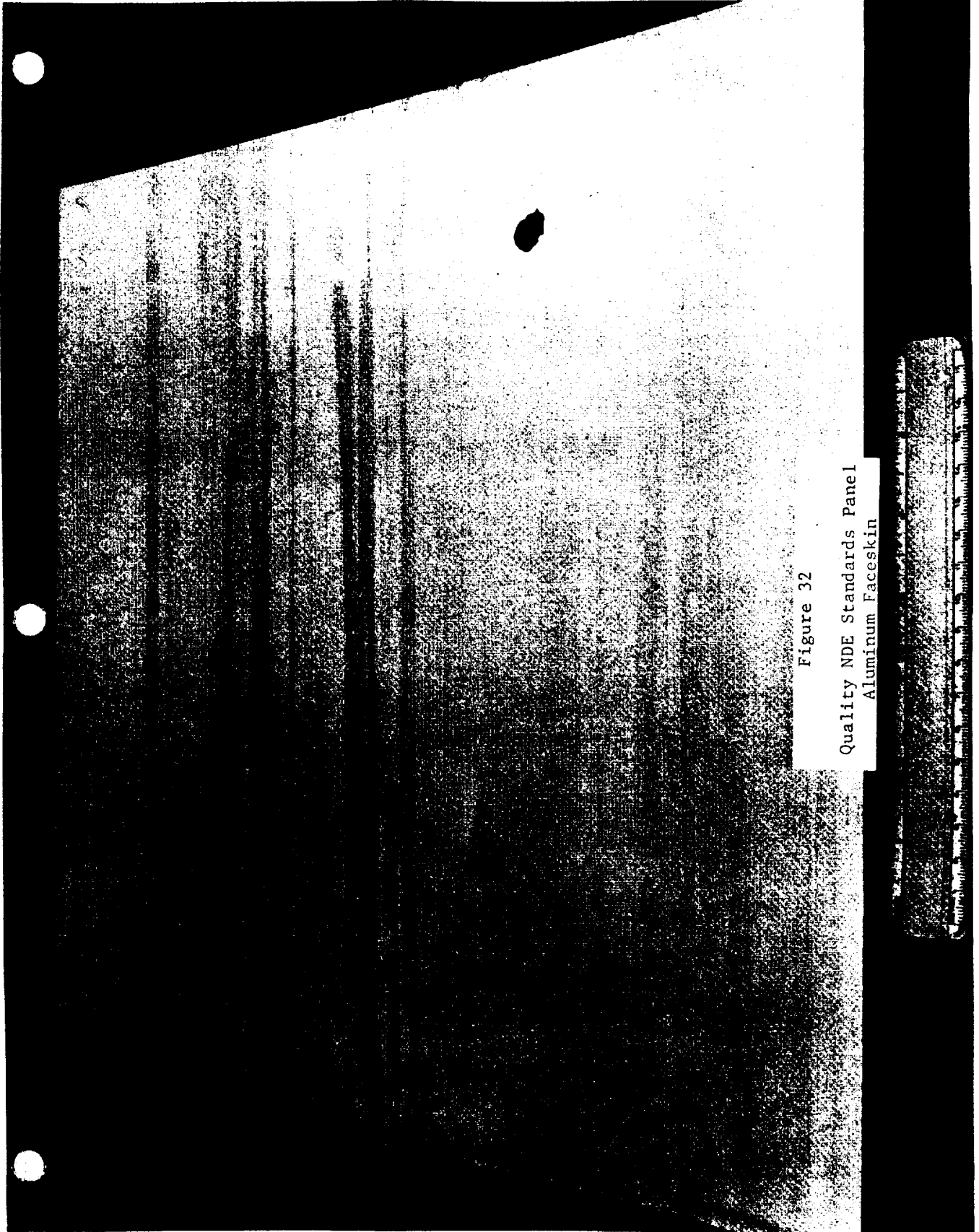


Figure 32

Quality NDE Standards Panel
Aluminum Faceskin

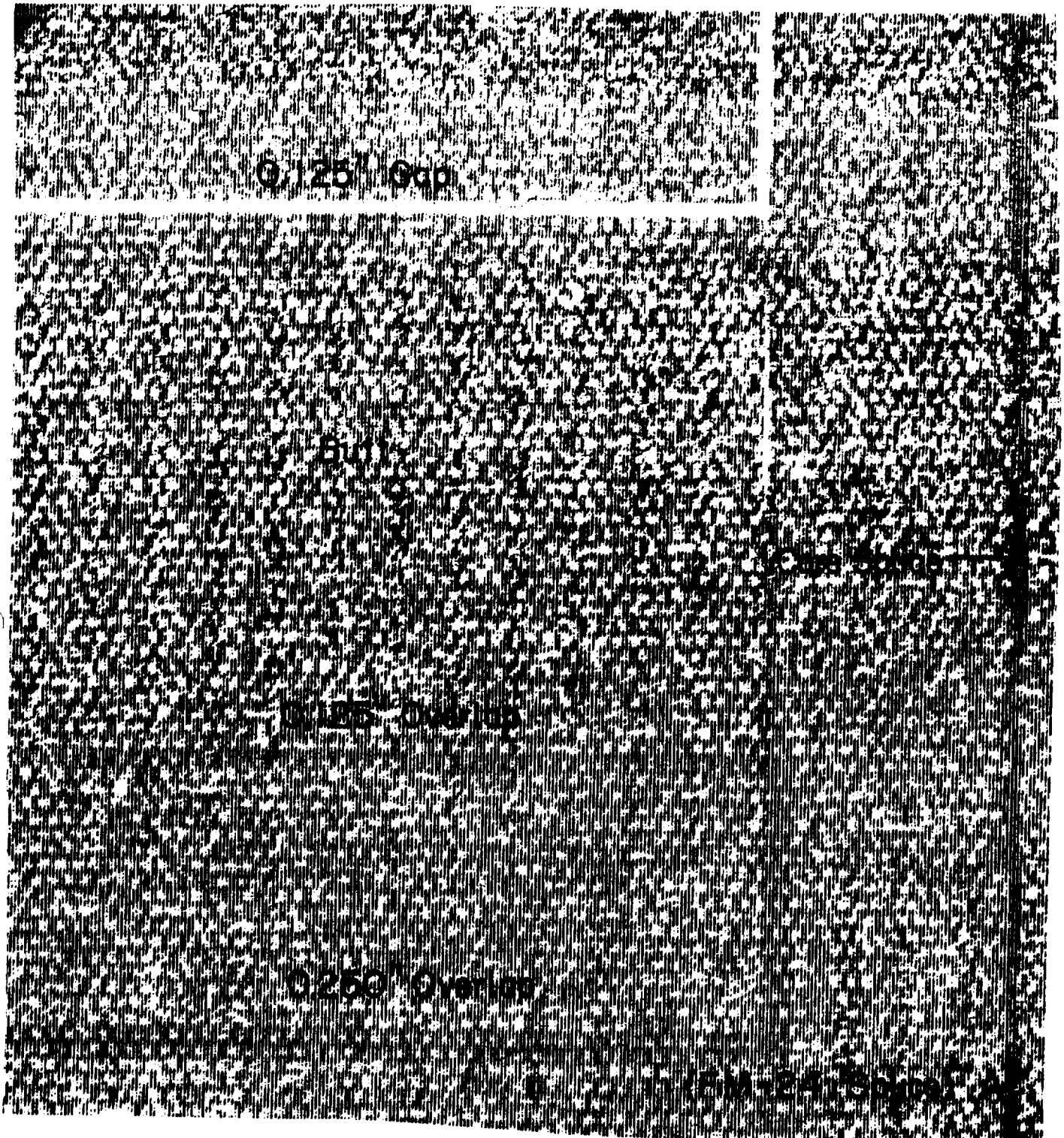


Figure 33
Ultrasonic C Scan
Aluminum Honeycomb Sandwich Panel
Quadrant A - FM-24 Splice

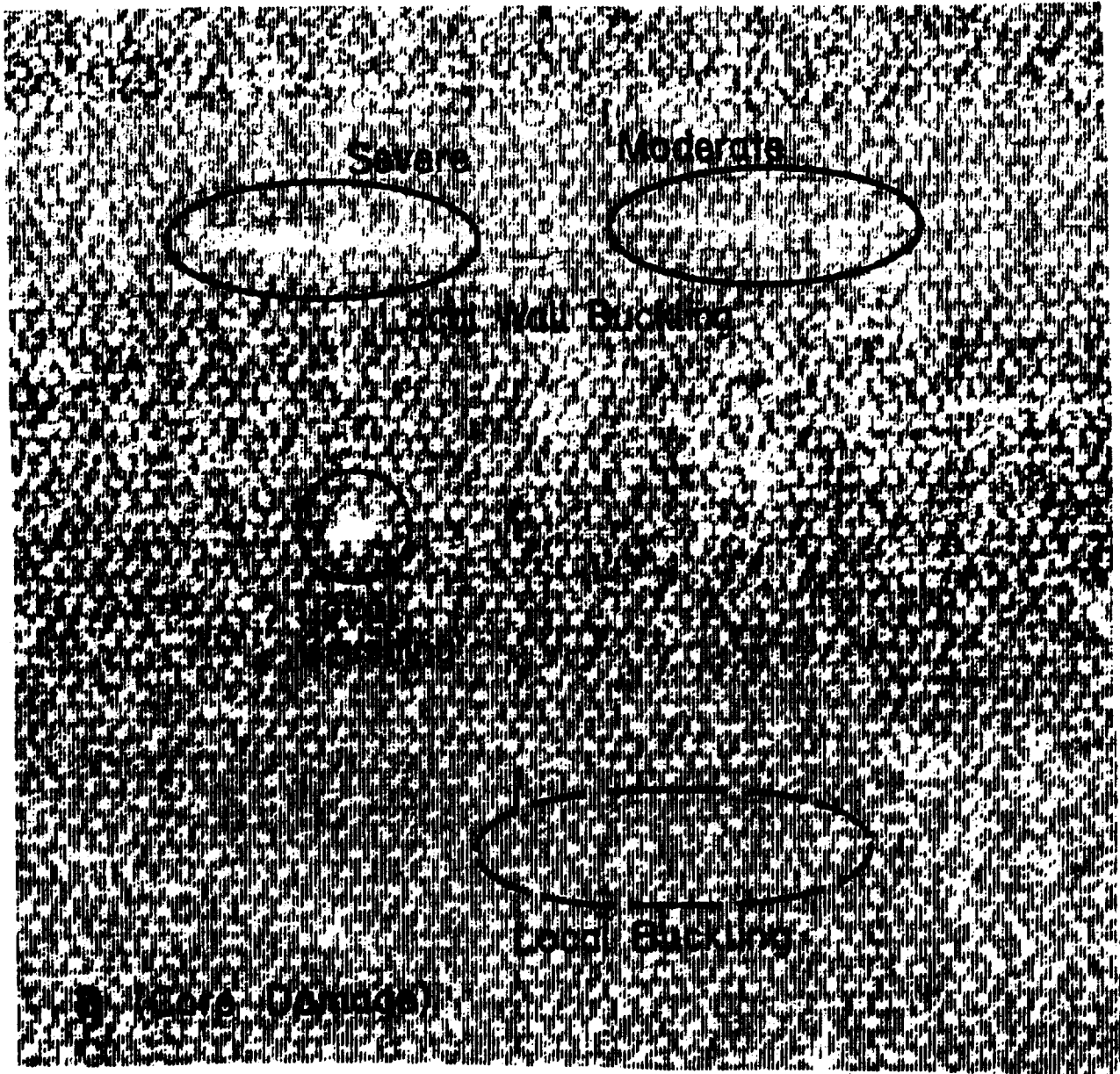


Figure 34

Ultrasonic C Scan
Aluminum Honeycomb Sandwich Panel
Quadrant B - Core Damage

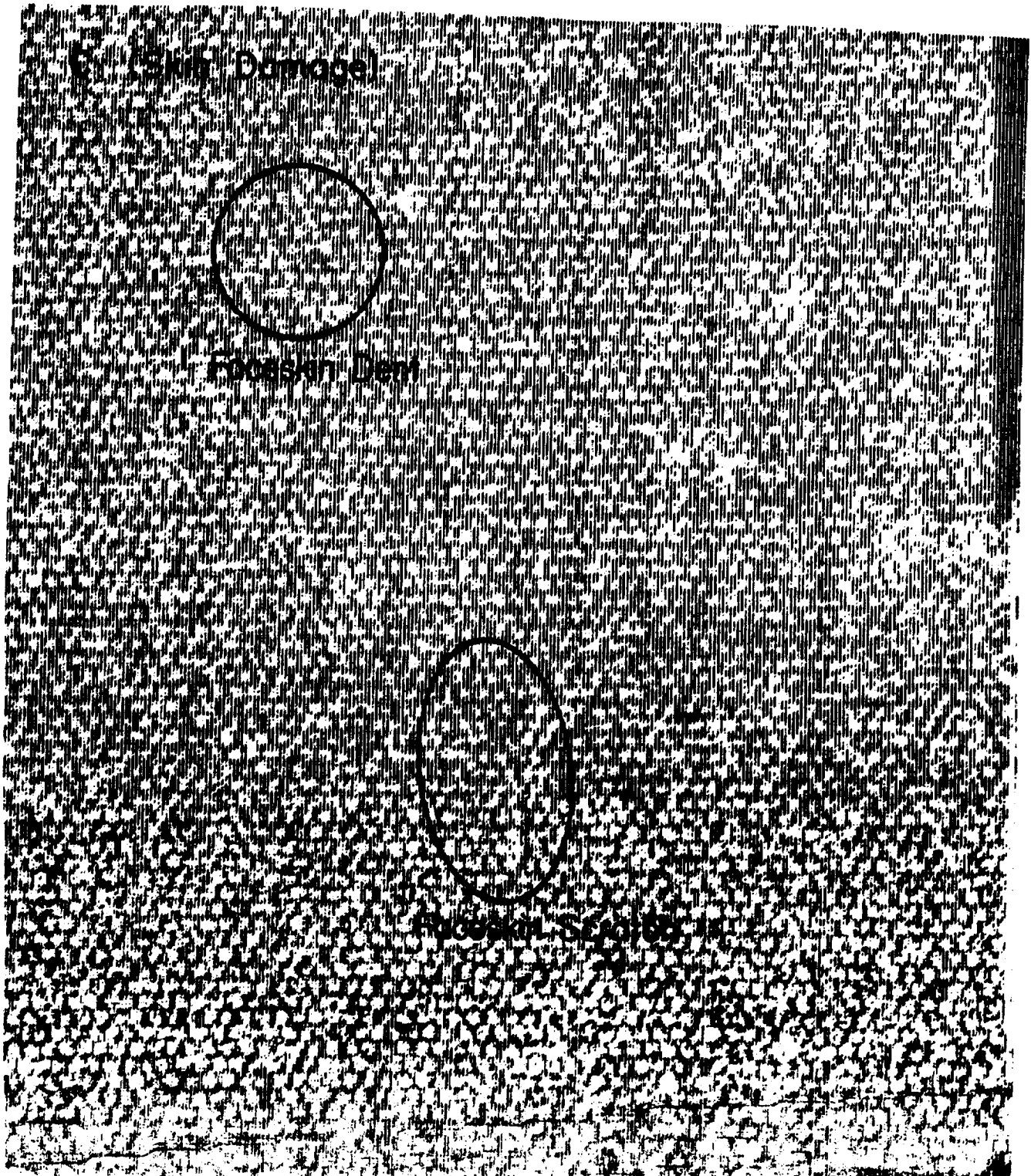


Figure 35

Ultrasonic C Scan

Aluminum Honeycomb Sandwich Panel

Quadrant B - Skin Damage

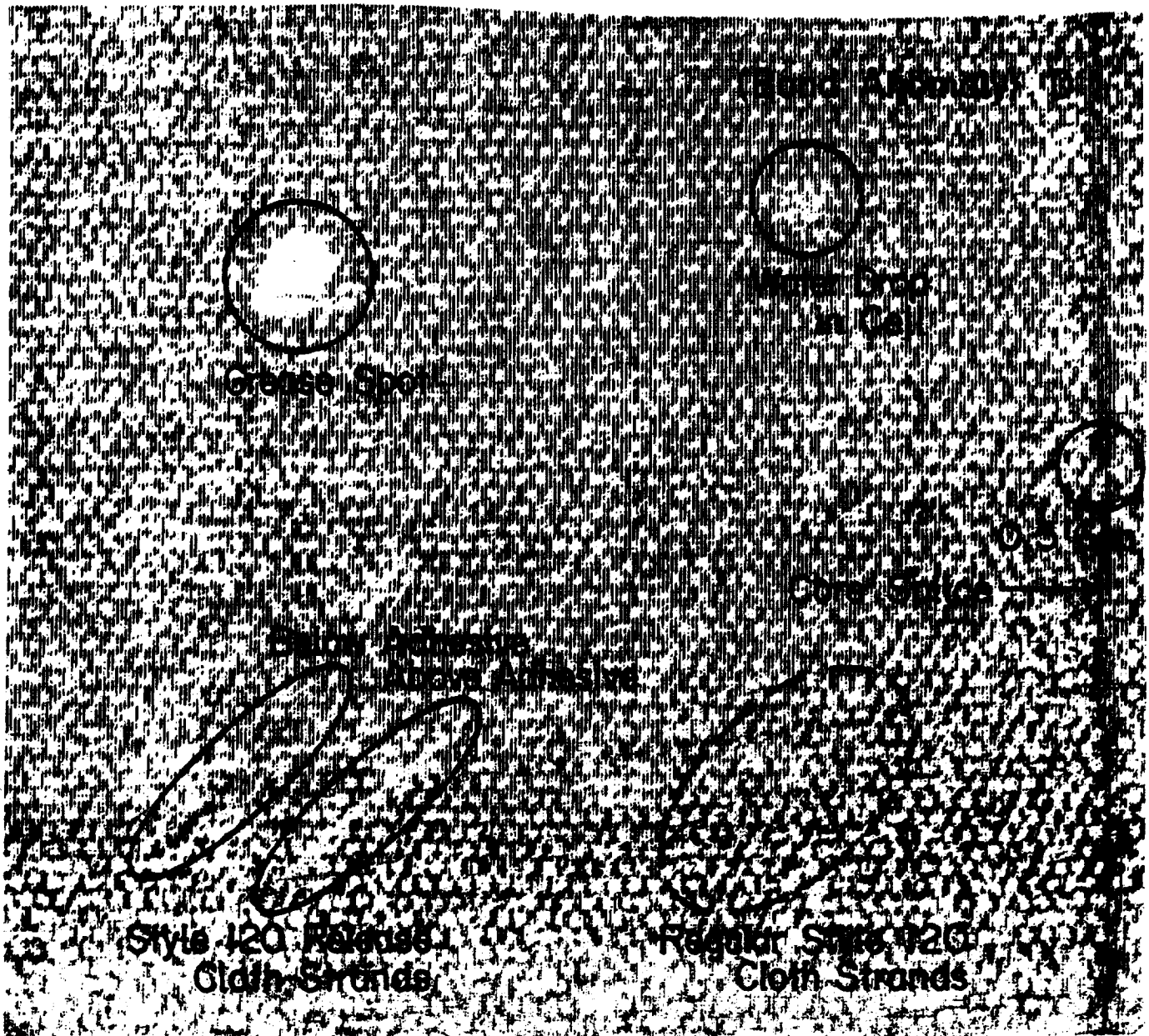


Figure 36
Ultrasonic C Scan
Aluminum Honeycomb Sandwich Panel
Quadrant D - Bond Anomaly

Local Wall Buckling

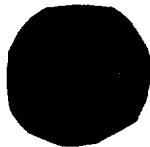
Severe



Moderate



Local
Crushing



Local Buckling



B - Core Damage

Figure 37
X-Ray
Aluminum Honeycomb Sandwich Panel
Quadrant B - Core Damage

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F. HOLBOAT Modification

The computer program "HOLBOAT" Ref. (3) calculates buckling loads of inhomogeneous anisotropic cylinders under combined loads. It is based on the Kirchhoff-Love hypothesis, generally anisotropic constitutive equations, and Flugge's differential equations of equilibrium. It was developed under contract to AFFDL and has been improved since then (circa 1967).

Cheng and Ho (Ref. 4) developed the basic equations for buckling by pressure, axial load and torsion. Their analysis was extended in Ref. 5 to include bending. Thus, any combination of pressure, axial load, torsion, and bending can be analyzed with the program and theoretical interactions determined.

The inhomogeneity considered is that which arises in a laminated cylinder due to different layers having different elastic properties and/or orientations. The elastic properties of each layer are input, along with its orientation and thickness. Then the program internally calculates the required shell stiffness. Each individual layer may be isotropic, orthotropic, or generally anisotropic and a symmetric or balanced arrangement of layers is not required.

Simple support boundary conditions are satisfied for "specially orthotropic" cylinders. For generally anisotropic configurations, no homogeneous boundary conditions are satisfied on sections perpendicular to the axis. If the cylinder is long or has a small axial stiffness, then these constraints will not greatly affect the buckling loads. However, short cylinders and those with high axial stiffness may be affected by boundary constraints.

Input to the program is via "Namelist". This means that the user does not have to have his input in "Format" but merely writes the name of the input variable, an equal sign, and the numerical value of the variable. This input may be in any sequence. The program can also run multiple problems and the user has only to input values of variables which changed from the previous problem. This feature is most useful in performing parametric studies. Program input consists of cylinder geometry, elastic properties of each layer, load combinations, wave number ranges, and buckling load type.

Ref. 3 - Holston, Jr., A., Feldman, A. and Stang, D. A.: Stability of Filament-Wound Cylinders Under Combined Loading. AFFDL-TR-67-55. Martin Marietta Corporation, Denver, Colorado, May 1967.

Ref. 4 - Cheng, S. and Ho, B. P. C.: Stability of Heterogeneous Anisotropic Cylindrical Shells Under Combined Loading. AIAA Journal, Vol. 1, No. 4, April 1963, pp 892-898.

Ref. 5 - Holston, Jr., A.: Buckling of Inhomogeneous Anisotropic Cylindrical Shells by Bending. AIAA Journal, Vol. 6, No. 10, October 1968, pp 1837-1841.

Output is shell stiffness, buckling load, and buckling mode shape. Two buckling loads are given for each wave number set; one based on a "Flügge type" theory and the second from a Donnell theory. The minimum buckling load and corresponding wave numbers are also printed for each data set.

Program Modifications

Two major improvements are being made to HOLBOAT that will enhance the program usefulness in obtaining more efficiently designed structures and free the user from performing some tedious input calculations. The areas being modified at this time are: 1) extension to stiffened cylinders and 2) incorporation of "knock-down" or reduction factors to obtain critical design loads.

1) Stiffened Cylinders

A cylinder with closely space stiffeners, inside, outside, or both, may be treated by "smearing" the stiffeners into an anisotropic sheet in the analysis. In this technique one determines a set of average stiffnesses for the stiffened cylinder and then determines buckling loads based on the average stiffnesses. The resulting buckling wave lengths are then compared with stiffener spacing to verify the smearing assumption.

The composition of the vertical stringers and circumferential frames are given in the modified program as "specially orthotropic". This permits the designer the option of using laminated or composite stiffeners as well as isotropic material stiffeners. (The present HOLBOAT program considers the skin as a generally anisotropic laminate.)

As for configuration, numerous geometries are available for use in reinforcing and providing stabilization to the structural skin of the cylinder. Seven of those most generally used, which are being incorporated into the program, are shown in Figure 38. These configurations can be used for both the vertical stiffeners and circumferential frames. The program user will have the option of choosing stringers, frames, configurations and inside or outside location. Each stiffener configuration has its advantages and disadvantages from structural, cost, fabrication, assembly, etc., standpoints. The variety shown will provide the designer a wide choice in his selections of particular configurations for evaluation.

2) Correlation Factors for Design Loads

A design buckling load is obtained by multiplying the calculated theoretical buckling load by a correlation factor. These factors are obtained from previously obtained test data and correlation studies and they reflect differences between the theory and test. Both initial imperfections and boundary conditions have been shown to be significant in causing these discrepancies. Most test data is not specific with regard to imperfections or boundary conditions; thus, the data from similar specimens and loadings are usually combined. Lower bound and/or statistical correlation curves are then drawn to provide the correlation factor.

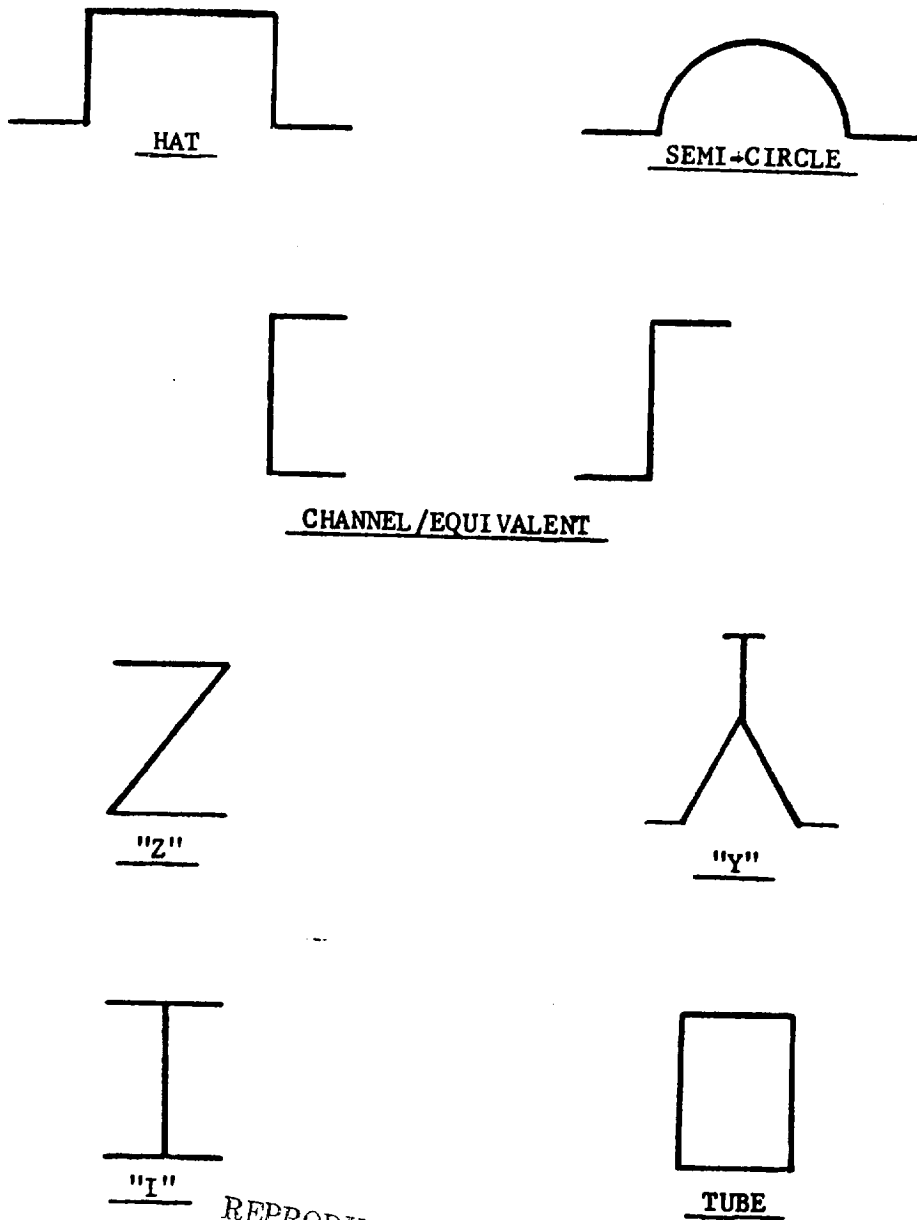
Correlation curves from the MSFC Astronautic Structures Manual (Ref. 6), the NASA cylinder buckling monograph, SP-8007 (Ref. 7) and engineering journals are presently being incorporated into the computer program.

The program user will choose the most appropriate curve and design buckling loads will be printed out together with the critical "classical" buckling load.

Following completion of the HOLBOAT program modification a User's Manual will be prepared. The manual will be written from the standpoint of the user being familiar with composites technology nomenclature and theory, shell instability theory, and Fortran programming. The manual will define input parameters and guide the user in selecting appropriate design assumptions. Diagrams will be presented to guide the user in setting up his design problem for evaluation. Inputs to the program will include structure geometry, design assumptions, and material properties. Output from the program will include critical "classical" buckling load (including interaction effects), critical design load, and buckled mode shape.

Ref. 6 - Astronautic Structures Manual, National Aeronautics and Space Administration, Marshall Space Flight Center, August 15, 1970.

Ref. 7 - Buckling of Thin-Walled Circular Cylinders. NASA SP-8007; NASA Space Vehicle Design Criteria (Structures), August 1968.



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FIGURE 38
CANDIDATE STIFFENER/FRAE GEOMETRIES

III. Schedule and Plan for Future Work

The master schedule shown in Figure 39 outlines the portion of work completed and major program milestones. The following tasks are scheduled for completion in the following quarter:

- 1) Complete HOLBOAT modification.
- 2) Complete User's Manual for modified HOLBOAT program.
- 3) Present one day seminar at NASA-MSFC on use of HOLBOAT.
- 4) Complete Phase II Structural Test Report.

DESIGN, FAB & TEST OF LIGHTWEIGHT
CELL STRUCTURE (NAS8-29979)

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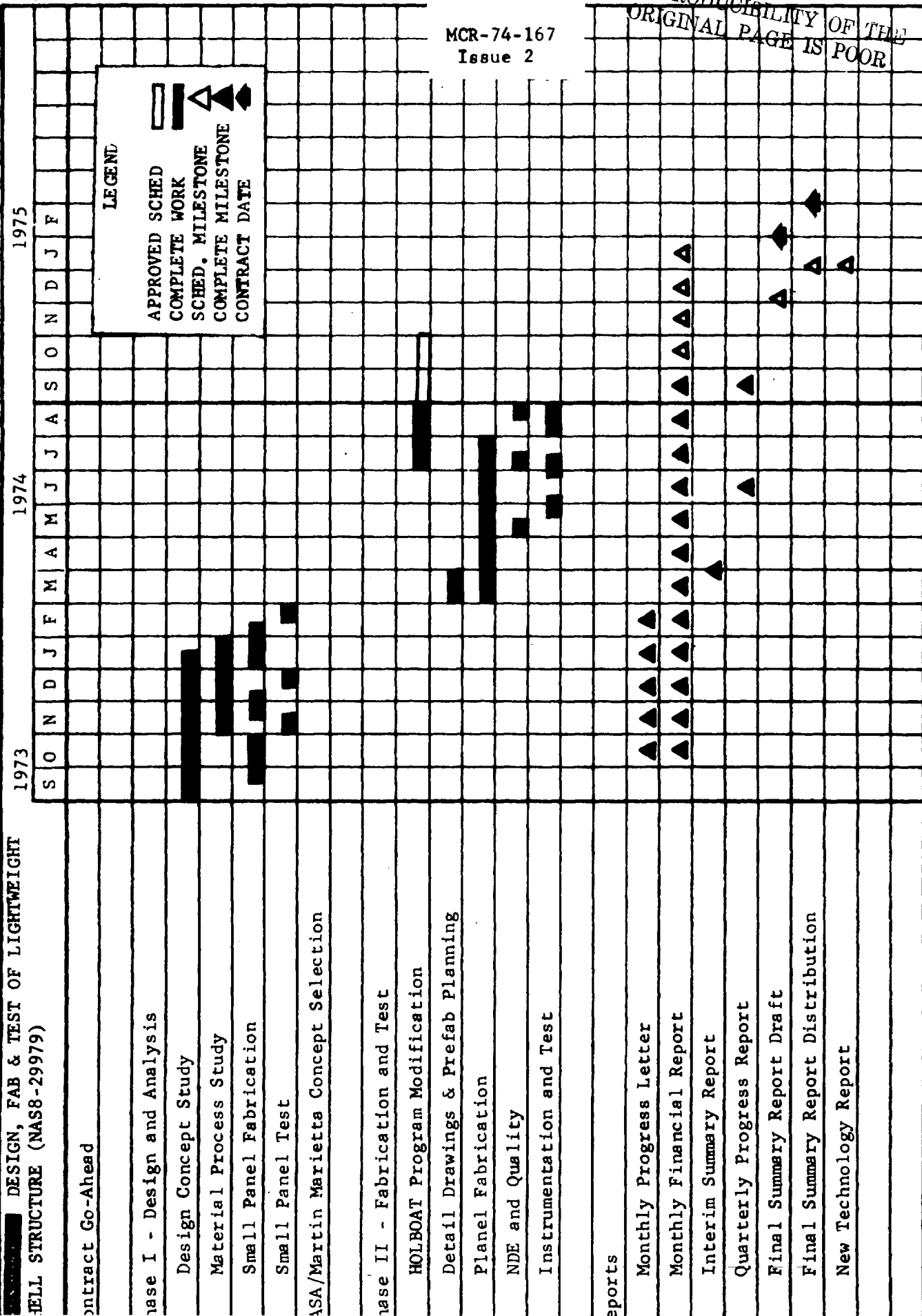


Figure 39. Program Master Schedule

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